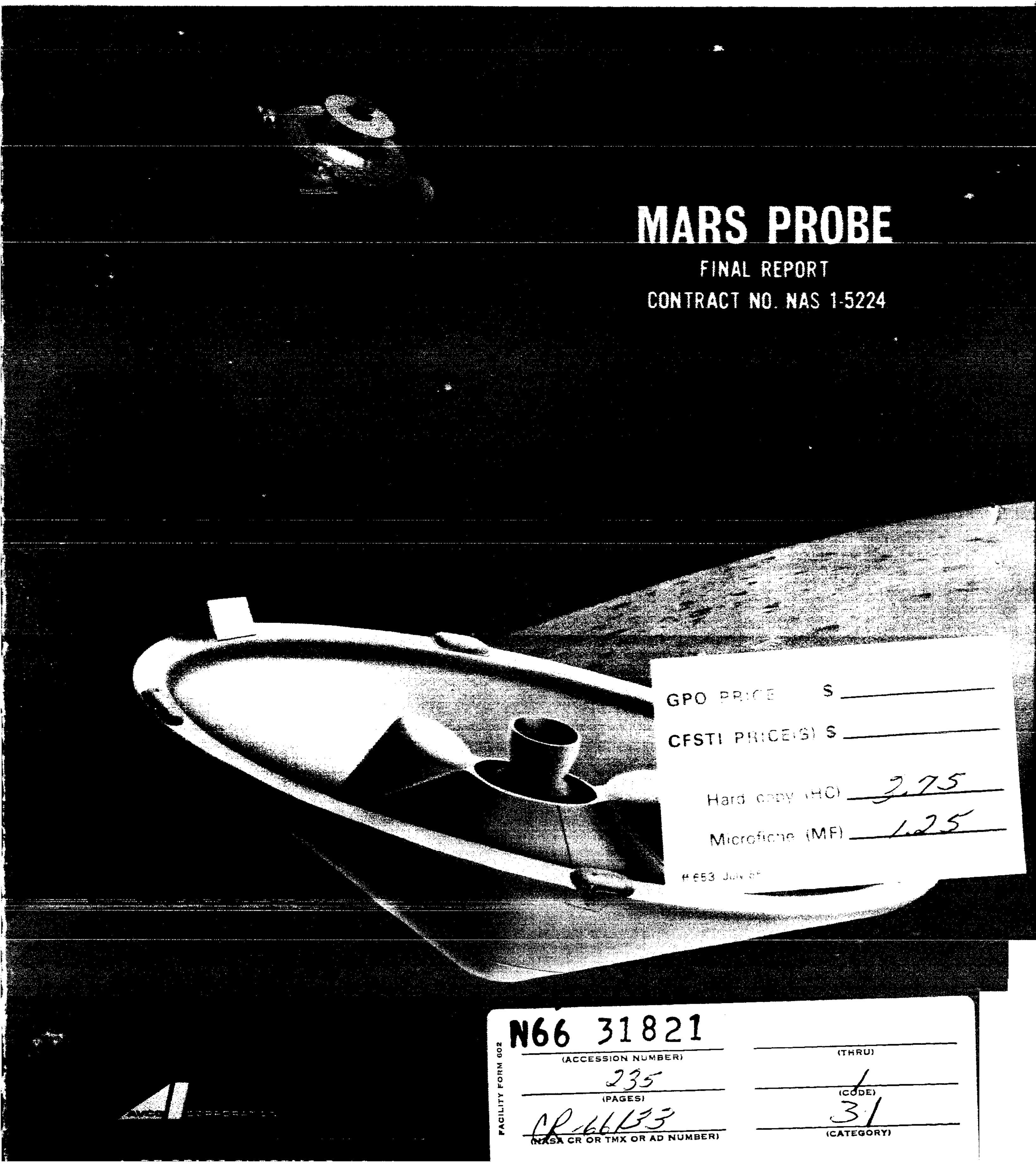
NASA CR-66133

VOLUME III — PROBE, ENTRY FROM ORBIT

BOOK 3

DEVELOPMENT TEST PROGRAMS



MARS PROBE

FINAL REPORT

CONTRACT NO. NAS 1-5224

BOOK INDEX

NASA CR-66133

COMPARATIVE STUDIES OF CONCEPTUAL DESIGN AND QUALIFICATION PROCEDURES FOR A MARS PROBE/LANDER

FINAL REPORT

VOLUME III PROBE, ENTRY FROM ORBIT

Book 3 DEVELOPMENT TEST PROGRAMS

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PREFACE

The results of Mars Probe/Lander studies, conducted over a 10-month period for Langley Research Center, NASA, are presented in detail in this report. Under the original contract work statement, studies were directed toward a direct entry mission concept, consistent with the use of the Saturn IB-Centaur Launch Vehicle, wherein the landing capsule is separated from the spacecraft on the interplanetary approach trajectory, some 10 to 12 days before planet encounter. The primary objectives of this mission were atmospheric sampling by the probe/lander during entry and terrain and atmosphere physical composition measurement for a period of about 1 day after landing.

Studies for this mission were predicated on the assumption that the atmosphere of Mars could be described as being within the range specified by, NASA Mars Model Atmospheres 1, 2, 3 and a Terminal Descent Atmosphere of the document NASA TM-D2525. These models describe the surface pressure as being between 10 and 40 mb. For this surface pressure range a payload of moderate size can be landed on the planet's surface if the entry angle is restricted to be less than about 45 degrees.

Midway during the course of the study, it was discovered by Mariner IV that the pressure at the surface of the planet is in the 4 to 10 mb range, a range much lower than previously thought to be the case. The results of the study were re-examined at this point. It was found that retention of the direct entry mission mode would require much shallower entry angles to achieve the same payloads previously attained at the higher entry angles of the higher surface pressure model atmospheres. The achievement of shallow entry angles (on the order of 20 degrees), in turn, required sophisticated capsule terminal guidance, and a sizeable capsule propulsion system to apply a velocity correction close to the planet, after the final terminal navigation measurements.

Faced with these facts, NASA/LRC decided that the direct entry from the approach trajectory mission mode should be compared with the entry from orbit mode under the assumption that the Saturn 5 Launch Vehicle would be available. Entry of the flight capsule from orbit allows the shallow angle entry (together with low entry velocity) necessary to permit higher values of M/C_DA, and hence entry weight in the attenuated atmosphere.

It was also decided by LRC to eliminate the landing portion of the mission in favor of a descent payload having greater data-gathering capacity, including television and penetrometers. In both the direct entry and the entry from orbit cases, ballistic atmospheric retardation was the only retardation means considered as apecifically required by the contract work statement.

Four months had elapsed at the time the study ground rules were changed. After this point the study continued for an additional five months, during which

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period a new design for the substantially changed conditions was evolved. For this design, qualification test programs for selected subsystems were studied. Sterilization studies were included in the program from the start and, based on the development of a fundamental approach to the sterilization problem, these efforts were expanded in the second half of the study.

The organization of this report reflects the circumstance that two essentially different mission modes were studied -- the first being the entry from the approach trajectory mission mode and the other being the entry from orbit mission mode -- from which two designs were evolved. The report organization is as follows:

Volume I, Summary, summarizes the entire study for both mission modes.

Volume II reports on the results of the first part of the study. This volume is titled Probe/Lander, Entry from the Approach Trajectory. It is divided into two books, Book 1 and Book 2. Book 1 is titled System Design and presents a discursive summary of the entry from the approach trajectory system as it had evolved up to the point where the mission mode was changed. Book 2, titled Mission and System Specifications, presents, in formal fashion, specifications for the system. It should be understood, however, that the study for this mission mode was not carried through to completion and many of the design selections are subject to further tradeoff analysis.

Volume III is composed of three books which summarize the results of the entry from orbit studies. Books 1 and 2 are organized in the same fashion as the books of Volume II, except that Book 2 of Volume III presents component specifications as well. Book 3 is titled Development Test Programs and presents, for selected subsystems, a discussion of technology status, test requirements and plans. This Book is intended to satisfy the study and reporting requirements concerning qualification studies, but the selected title is believed to describe more accurately the study emphasis desired by LRC.

Volume IV presents Sterilization results. This information is presented separately because of its potential utilization as a more fundamental reference document.

Volume V presents, in six separate books, Subsystem and Technical Analyses. In order (from Book 1 to Book 6) they are:

Trajectory Analysis
Aeromechanics and Thermal Control
Telecommunications, Radar Systems and Power
Instrumentation
Attitude Control and Propulsion
Mechanical Subsystems

Most of the books of Volume V are divided into separate discussions of the two mission modes. Table of Contents for each book clearly shows its organization.

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ACKNOWLEDGEMENT

The conduct of the study and technical preparation of this report involved the participation and close coordination of many people, all of whose contributions were important to the end results.

It is impractical to single out each individual, but the major contributors to the study, reflected in the material presented in this Book, are as follows:

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INTRODUCTION

Under the original contract work statement it was intended to develop qualification procedures and program planning for the selected probe/lander design. As pointed out in the Preface, the change in the Mars density-profile estimates, as occasioned by Mariner IV results, caused significant redirection of the study effort and a major change in the system design.

Redirection of emphasis and scope of the qualification-procedures portion of the study also took place at that time. Up to that point, preliminary study of qualification procedures and program planning had taken place for the first design - that of the probe/lander direct entry concept. With the change to the probe, entry from orbit design concept, the previous work on qualification procedures was in many respects negated, and future efforts were directed solely to the entry from orbit design. The results for the latter case are presented in this book.

Additionally, at the time of redirection, subsystem development status and critical development test programs were emphasized rather than the former emphasis on formal qualification procedures. The previous program planning requirements were also eliminated.

The redirection also excluded (as far as development test planning was concerned) consideration of certain subsystems such as the instrumentation, power, and instrumentation subsystems.

In summary, this book presents development status and test programs for selected subsystems of the probe, entry from orbit design.

1.0 STUDY REQUIREMENTS

This book (Book 3, Volume III) has been prepared to meet the requirements of paragraphs 4.3(2) and (3) of NASA/Langley Research Center Statement of Work L-5295c, Exhibit D, entitled, "Comparative Studies of Conceptual Design and Qualification Procedures for a Mars Probe/Lander" dated December 16, 1965. These paragraphs read as follows:

"4.3(2) Procedures, equipment and facilities shall be defined for the ground testing of those components and subsystems which are deemed to have critical development problems.

"4.3(3) The Contractor shall study the value and extent of flight tests in the Earth's atmosphere in the development of subsystems. Trajectory and launch details of Earth entry flight tests corresponding to Mars trajectories shall be identified for both scaled and prototype configurations insofar as environmental conditions are concerned. Degree of similitude achievable with respect to loadings, subsystem operations, and component actions shall be determined. The feasibility of checking out, during the flight tests in the Earth's atmosphere, the electrical, mechanical and communication interfaces with the Bus shall be determined."

Consistent with the guidelines given in Section 3.0 of the aforementioned Statement of Work, the following selected portions of the Probe system were considered:

(1) Structure and heat shield (Probe shell)

(2) Sterilization canister

(3) Probe-Bus separation system

(4) Attitude control system

(5) Propulsion system, and

(6) Parachute system

Chapters 2 through 7 deal with development ground testing, whereas Chapters 8 through 10 deal with the recommended flight tests. An Appendix is included which presents a preliminary program plan for the 1971 Probe/Lander mission. This plan was developed prior to the NASA/LRC redirection of the study effort and, although no longer directly applicable, has been included for the sake of completeness.

SUBSYSTEM DEVELOPMENT STATUS AND CRITICAL GROUND DEVELOPMENT TESTS

2.0 ENTRY VEHICLE SHELL SUBSYSTEM

The performance of the entry vehicle depends on a number of closely interacting environmental, structural and material dependent factors. Although the shell elements perform various functional tasks in flight they are exposed to the same environments. Thus it is desirable to plan the development test program in such a way that as many tests as possible are integrated to provide design or performance prediction information for more than one element (component) of the system or technological discipline.

In the development of the entry-vehicle shell, it is necessary to establish:

1. The vehicle aerodynamic performance (coefficients) consistent with the anticipated mission requirements (on-board experiments, communications, payload) and flight profile.

2. The aerothermodynamic environment (heating, loads, pressures).

3. The response of the shell to the environments in terms of the structural, thermal protection, and thermal control behavior.

4. Manufacturing methods and concepts.

The developmental tests should be conducted in facilities which are capable of closely reproducing the environmental levels. One of the main tasks in the planning of development tests is the determination of the degree of simulation required and the selection of facilities consistent with the time schedule allocated for the program.

Wherever possible recommended tests have been integrated to achieve combined objectives. For example, combined tests are proposed to determine aerodynamic performance and environments; aero environments and thermal response in aggravation areas; thermo-structural behavior and thermal control requirements during sterilization and spaceflight; and behavior of heat shield and thermal control coating, to mention a few.

No excessive extension of the available technology was found; however, critical problem areas do exist. In some cases significant lead times are required to provide the necessary information or to develop fabrication and test techniques on time. In other cases there is a lack of basic information. Among those areas are, for example:

1. An afterbody geometry to provide one stable trim point.

2. The uncertainty in the effect of postulated Mars atmospheres on the ablative behavior of materials.

3. The stability of honeycomb sandwich conical shells with "weak" boundary conditions.

4. The unpredictability of thermal control performance when a complicated system of joints, and conductive and radiative paths is investigated.

2.1 AEROTHERMODYNAMICS

2.1.1 Reference Design Performance and Technology Development Requirements

Aerothermodynamic analyses provide the environment in terms of the imposed thermal and structural loads as well as the vehicle stability and performance. This involves determining pressure and heating distributions and aerodynamic coefficients. The development testing should be aimed at filling basic information gaps and investigating critical areas.

The velocities associated with entry out of orbit are such that radiative heating does not contribute significantly to the environments; thus only convective heating need be investigated. A significant reduction in the development test program can be realized if the ground tests are restricted in the extent to which atmospheric composition is varied. Considerable data have already been obtained on the effects of atmospheric composition on the convective heating. Thus, it is recommended that the ground tests be conducted on the reference configuration in air with the data presently available being utilized to account for composition effects.

In particular, the desired information should be established under real gas conditions, the relevant parameter in this case being the stagnation point density ratio, ρ_s / ρ_∞, which is a measure of the effective specific heat ratio as well as the shock standoff distance. The simulation of ρ_s / ρ_∞ is necessary to ensure adequate determination of the performance and environments.

The aerothermodynamic testing has been divided into three elements: (1) the afterbody, (2) the forebody, and (3) the entry configuration comprising the afterbody and forebody.

The afterbody development is critical in terms of the overall system requirement. Its primary function of ensuring only one stable trim point can result in significant penalties not only in weight but in terms of other system interfaces such as the ΔV-rocket location. The early phase of the program would determine if a minimum afterbody be justified and if auxiliary destabilizing devices such as asymmetries or flaps be needed.

Primary emphasis for the forebody is on the generation of basic design information, such as pressure distributions and heating distributions. Included in these tests are the effects of protuberances and cavities, which will be examined on the reference configurations to ensure the proper local flow environments and obviate the need for possible parametric studies.

The configuration performance and stability development will require a complete Mach No. variation as well as testing in a gas other than air to determine the possible effects of density ratio on the vehicle aerodynamic coefficients.

Tables I and II summarize the aerothermodynamic development requirements and tests. The simulation requirements are shown in Figure 1, where flight conditions at various critical phases are delineated.

2.1.2 Afterbody Tests

2.1.2.1 Test Objectives and Description

The existing data relevant to rearward stability indicates possible problem areas; the data, however, include both sting and forebody contributions which cannot be factored out. The afterbody can contribute significantly to the shell weight both structurally and thermally thus necessitating ground tests to establish its performance (particularly the rearward stability) and environments. These characteristics are of significance at low Reynolds Numbers (early entry). The need for and the effectiveness of rearward destabilizing mechanisms (flaps, asymmetries, etc.) should also be established. All anticipated protuberances and cavities should be investigated to determine the local heating aggravations. The requirements in terms of configuration and/or modifications to ensure one stable trim point should also be defined.

The simulation of the critical flight parameters (low Reynolds Number and high Mach No.) does not present a problem; however, the desire to realize the anticipated density ratios simultaneously will require ballistic range tests in addition to wind-tunnel tests.

Additional problems are associated with the method of model support. Although sting effects are normally alleviated by minimizing the sting diameter, it is felt that the sting still provides an attachment point for the near wake rendering the test data suspect, especially since

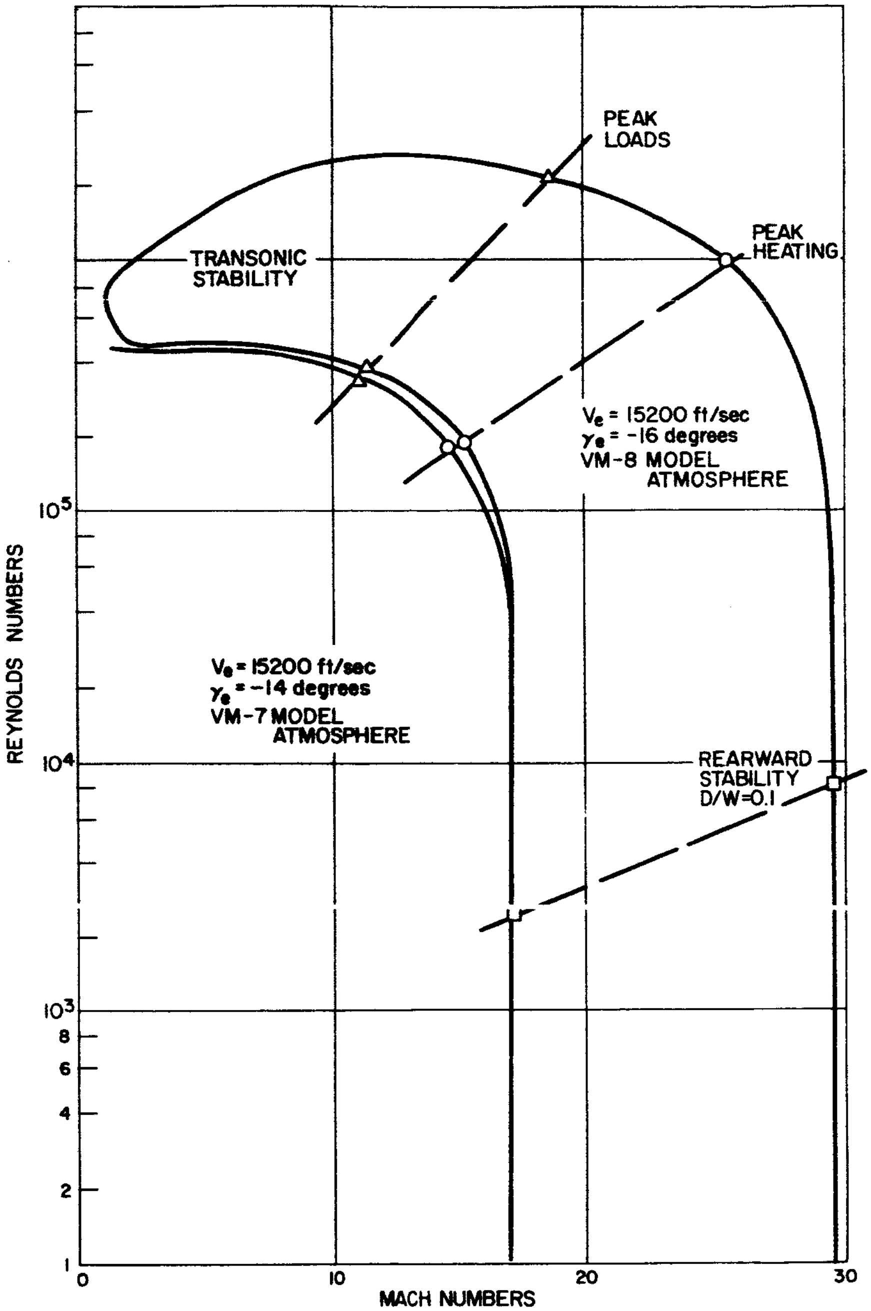

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Figure 1 MARS ENTRY SIMULATION REQUIREMENTS

TABLE I

AEROTHERMODYNAMIC TEST SYNTHESIS SUMMARY

Item No.	Function (Primary)	Critical Area	Description of Elements, Subassembly, etc.	Phase of Mission	Design Information	Performance Prediction	Flight Test Design Verification Required
1	Convective Heating	Yes	Afterbody	Early Entry	X		
2	Stability	Very	Afterbody	Early Entry	X	X	
3	Convective Heating	Yes	Forebody	Entry	X		Yes*
4	Loads	Yes	Forebody	Entry	X		Yes*
5	Performance and Stability	Yes Very	Entry Configuration	Throughout Entry		X	Yes*
6	Stability		Parachute Flight Test Configuration	Post Entry		X	

*Note: These verifications will be conducted only in conjunction with the heat shield flight test.

TABLE II

AERODYNAMIC DEVELOPMENT

	Element	Mission Phase of Concern	Design Analysis Problem Area or Requirement	Test Objectives	Test Description	Desired Test Conditions	Typical Test Facilities
(A)	Afterbody	Early entry and entry	Heating and stability	Determination of: 1. Rearward stability 2. Need for and effectiveness of flaps and/or asymmetries 3. Heating distributions and local aggravations	1. Static force and moment measurements in wind tunnels and ballistic range 2. Wind-tunnel measurements of heating and aggravations	1. Hypersonic Mach Nos. (greater than 10) 2. Low Reynolds Nos. (less than 10^5)	1. Ames Hypersonic Blowdown Tunnel 2. JPL Hypersonic 3. NOL Ballistic Range
(B)	Forebody	Entry	1. Heating 2. Pressure Distributions	Determination of: 1. Pressure/Distributions 2. Low Reynolds No. effects on heating 3. Angle of attack effect on heating and loads 4. Local aggravations	Wind-tunnel measurements of heating and pressures	1. Hypersonic Mach Nos. (greater than 10) 2. Reynolds Nos. between 10^4 and 10^6 3. Density ratios as high as possible	1. Ames Hypersonic Blowdown Tunnel 2. Cornell Wave Superheater 3. Jangley Hypersonic
(C)	Entry Configuration	Early entry Entry and post-entry	Performance and stability	Determination of: 1. Static and dynamic stability derivatives 2. Force derivatives 3. Composition effects	Force and Moment Measurements in Wind Tunnels and Ballistic Range	Complete simulation of Mach and Reynolds Nos. variations and density ratios extent as well as angle of attack	1. Eight-Foot Transonic Tunnel Langley 2. Unitary Wind Tunnel 3. Ames Hypersonic Blowdown Tunnel 4. NOL Ballistic Range 5. BRL Ballistic Range
(D)	Parachute Flight Test Vehicle	Post-entry	Transonic stability and performance	Determination of: 1. Static and dynamic stability derivatives 2. Force derivatives at transonic speeds	Force and Moment Measurements in Wind Tunnel	Transonic Mach. Nos. at moderate Reynolds Nos.	Eight-Foot Transonic Wind Tunnel Langley

the rearward stability for the candidate afterbody configurations is nearly neutral near 180 degrees angle of attack. The possibility of wire supported models or free-flight model testing is suggested.

The anticipated critical flight parameters for the afterbody (see Figure 1) occur at an axial deceleration of 0.1 g where the initial turn-around for rearward entry commences. The hypersonic Reynolds No. simulation requirement is seen to be between 10^3 and 10^4. Force measurements and heating distributions at angles of attack varying from 135 to 180 degrees should be obtained in this Reynolds No. range.

2.1.2.2 Facilities, Equipment and Test Conditions

The recommended facilities and test conditions are summarized in Table III. The rearward stability is determined by three independent tests:

1. Wind-Tunnel Tests

   a. Sting supported (Ames)

   b. Free flight (JPL)

2. Ballistic Range (NOL)

The primary emphasis should be on the Ames facility with the JPL and NOL data utilized to substantiate the results obtained with the sting supported models. In addition, the NOL data will indicate the effects of high density ratio upon the stability.

The heating distributions for the basic afterbody will be obtained by standard procedures (thin-film gage technique). It is recommended that for protuberances and cavities, thermal sensitive coatings be used to determine the local aggravations.

The sting mounted wind-tunnel tests will require three basic models: 1) Force measurements model, 2) Thin-film calorimeter model, and 3) Thermal sensitive coating model.

The free-flight models are small in size and simple design and it is anticipated that ten would be required for each facility (JPL and NOL).

TABLE III

AFTERBODY GROUND TESTS

Facility	Mach. No.	$R_{\infty D}$	α	Type Test
1. Ames Hypersonic Blowdown Tunnel	12	10^3 - 10^5	135 - 180 degrees	Static force, heat transfer in air
2. JPL Hypersonic (21 inch)	10	$\sim 10^4$	160 - 180 degrees	Static stability derivatives in air
3. NOL 300-Foot Ballistic Range	5 - 15	10^3 - 10^5	160 - 180 degrees	Static stability derivatives in air and freon. spark photographs

2.1.3 Forebody Tests

2.1.3.1 Test Objectives and Description

Although radiation heating is insignificant for the entry out of orbit mission, the shallow angles associated with this type of entry result in significant portions of the heat pulse at low Reynolds Numbers. Low-density effects (such as vorticity interaction) as well as entropy variation effects influence the heat pulse substantially. The latter effect arises from the growth of the boundary layer emanating from a region of lower entropy change through the bow shock.

Dynamic analyses indicate that during the heat pulse the entry vehicle is subjected to large variations in angle of attack, which significantly affect the thermal design. The stagnation point location during heating may be discontinuous, being on the spherical nosecap at nominal angles of attack (less than 30 degrees) and at the maximum diameter region at higher angles of attack, resulting in a significant alteration of the heating distribution. The heating in the latter case is dependent upon the local radius at the maximum diameter. To reduce the heating by increasing the radius would decrease the drag, an undesirable result.

The forebody development tests should be concerned primarily with the peak heating and peak loads phase of reentry as shown in Figure 1. The Reynolds and Mach Nos. present no simulation difficulties.

The forebody ground tests should establish:

1) Pressure distributions and dependence on density ratio

2) Stagnation point heating variation with low Reynolds Numbers (vorticity interaction).

3) Extent of entropy variation effects on heating.

4) Angle of attack effects on flow field and heating, and

5) Local heating aggravations associated with protuburances and cavities.

Although the Reynolds and Mach No. simulation presents no problem, the simultaneous satisfaction of the entropy variation along the boundary layer may be difficult. In addition, the simulation of the density ratio across the bow shock is necessary to obtain the correct velocity gradients, not only at the stagnation point, but at the sonic point as well.

The ground tests should provide Reynolds No. and angle of attack variation at hypersonic Mach Nos. consistent with those associated with the critical phases of entry. The Mach and Reynolds Numbers at both peak heating and loads are indicated in Figure 1 for the range of trajectories and atmospheres of interest. Furthermore, to obtain knowledge of the heat pulse, it is necessary that the Reynolds No. variation be $1 \times 10^4 < R_{\infty D} < 5 \times 10^5$. The pressure needs only limited investigation; the density ratio across the bow shock should be at least 10 with a sufficient variation to permit the establishing of the pressure distribution dependence. The angle of attack variation occuring at times of interest for structural design are within ± 30 degrees and ± 90 degrees for heating.

2.1.3.2 Facilities and Test Conditions

The recommended ground test program is summarized in Table IV giving both the facility and the test conditions. Since the simulation available is not ideal with respect to the realization of both the density ratio and the entropy variation, the test program will serve as substantiation. Sufficient variation in the relevant parameters will permit interpolation and/or extrapolation.

The Cornell Wave Superheater (which provides large density ratios) in conjunction with the Ames Hypersonic Tunnel tests will be used to establish the following: 1) Low density effects, 2) Density ratio effects, and 3) Angle of attack variations on heating and pressures.

The Ames and Cornell heat transfer measurements will be obtained by means of the thin-film calorimeter technique; the Langley tests will be conducted utilizing thermal sensitive coatings to establish the local aggravations due to protuberances and cavities.

The Cornell and Ames tests will require a minimum of two models each; one instrumented for pressures and the other for heating measurements. Additional models will be required due to the angle of attack variations desired. The Langley tests will require one basic model which can be modified for the protuberances and cavities; the number of models necessary will depend upon the ability to recoat and reuse.

2.1.4 Entry-Vehicle Shell (Forebody-Afterbody Unit)

2.1.4.1 Test Objectives and Description

In order to conduct the trajectory analyses, the complete vehicle performance characteristics are necessary. These are evaluated in

TABLE IV

FOREBODY GROUND TEST PROGRAM

Facility	Mach No.	Reynolds No. ($R_{\infty D}$)	Angle of Attack α	Type Test
1. Cornell Wave Superheater	10 - 18	$10^4 - 10^6$	0 - 50 degrees	Pressure and heating
2. Ames Hypersonic Wind Tunnel	15	$10^4 - 10^6$	0 - 90 degrees	Pressure and heating
3. Langley Mach. 8 Hypersonic	8	$10^4 - 10^6$	0 - 50 degrees	Heating

terms of the aerodynamic coefficients which are functionally dependent Mach No. and Reynolds No. and density ratio, the latter of which may be used to reflect real gas effects.

Figure 1 shows the Mach No. - Reynolds No. regimes experienced during entry. Table V summarizes the critical phases and the attendant flight environments.

The early entry performance characteristics were previously discussed (afterbody - see Table II). The entry performance characteristics are necessary to ensure adequate convergence during the critical phases of flight (peak heating, loads, etc.). The primary coefficients of interest are the static forces and moment derivatives. Only a cursory look at the dynamic derivatives at hypersonic speeds is necessary but the investigation should include the effects of mass injection. The post-entry performance will receive primary emphasis and will include extensive dynamic testing to ensure that transonic divergence, if it exists, is within tolerable limits; the existence of limit cycles would also be determined.

The aerodynamic coefficients C_x, C_N, C_m and C_{m_q} should be established as functions of Mach No. density ratio, angle of attack and, to a lesser degree, Reynolds No.

The desired ranges of these parameters are given in Table V. The hypersonic performance should be evaluated with a density ratio between 10 and 13 to 1 ($10 < \rho_s / \rho_\infty < 13$) to evaluate the real gas effects.

2.1.4.2 Facilities Equipment and Test Conditions

The recommended facilities and test conditions are tabulated in Table VI. Extensive angle of attack variations are limited to the hypersonic tests (Ames Blowdown). All the facilities have provisions for measuring the dynamic characteristics, however forced oscillation tests should be conducted in the supersonic and transonic facilities (Langley Transonic and Unitary Wind Tunnels). In addition to the angle of attack coverage in the Ames tests, some testing at varying Reynolds No. is recommended.

The ballistic range tests would be used to establish the dynamic characteristics in conjunction with the forced oscillation tests. In addition, the density ratio effects on both the static and dynamic derivatives may be obtained by means of ballistic range shots into Freon gas. This gas has the advantage of a low sonic velocity (in addition to the higher density ratios) thereby lowering the required velocities for a given Mach No.

TABLE V

ENTRY VEHICLE SHELL GROUND TESTS

Phase	Mach. No.	Reynolds No.	Angle of Attack	Critical Area
1. Early entry	17 - 30	$R_{\infty D} < 10^4$	$\alpha \sim 180°$	Rearward stability
2. Entry	6 - 30	$10^4 < R_{\infty D} < 3 \times 10^6$	$\alpha < \sim 150°$	Angle of attack convergence
3. Post-entry				
a) Supersonic	$1.5 < M_\infty < 6$	$4 \times 10^5 < R_{\infty D} < 2 \times 10^6$	$\alpha < 20°$	Static and Dynamic Stability
b) Transonic	$.8 < M_\infty < 1.5$	$4 \times 10^5 < R_{\infty D} < 10^6$	$\alpha < 50°$	
c) Subsonic	$0 < M_\infty < .8$	$4 \times 10^5 < R_{\infty D} < 10^6$	$\alpha < 50°$	

TABLE VI

ENTRY VEHICLE SHELL SUBSYSTEM DEVELOPMENT

Facility	Mach No.	Reynolds No.	Angle of Attack	Type Test
1. 8-foot Transonic Tunnel NASA Langley	$.6 < M_\infty < 1.2$	[illegible] $< R_{\infty_D} < 1.3 \times 10^6$	$0 < \alpha < 50°$	Static and Dynamic Stability Forced Oscillation
2. Unitary Wind Tunnel	$1.5 < M_\infty < 5.0$	2×10^6	$0 < \alpha < 50°$	Static and Dynamic Stability Free and Forced Oscillation
3. Ames Hypersonic Blowdown Tunnel	12	$10^{[illegible]} < R_{\infty_D} < 2 \times 10^6$	$0 < \alpha < 150°$	Static Forces and Stability (Including Ablating Model)
4. NOL 300-foot Ballistic Range	5 - 15	10^6	$0 < \alpha < 10°$	Drag and Static Stability (Including Freon Tests)
5. BRL Ballistic Range	$.8 < M_\infty < 5.0$	10^6	$0 < \alpha < 10°$	Drag, Static and Dynamic Stability

Each wind-tunnel facility will require one model except the Ames tests where two models would be a minimum to obtain the required angle of attack variations.

The ballistic range requirements depend upon the recoverability of the individual shots. In the case of the transonic tests low inertia models are required (to increase the number of oscillations per shot). Molded plastic foam models offer a possible means for achieving low inertia; however their reuse is precluded.

2.2 THERMAL PROTECTION

2.2.1 Reference Design Performance and Technology Development Requirements

The thermal protection system (TPS) consists of the composite of an external layer of heat shielding material bonded to the load carrying structure. The critical problems which arise in the development cycle are the feasibility of a concept or practicability of meeting the overall weight limitations (allocations) of the system within the allocated time schedule.

The practicability of meeting the schedule deadlines manifests itself at several stages of the development program and is affected by several factors: a) availability of the heat shield material (ablator); b) availability of the bonding material; c) system design information required prior to the design freeze (properties and material characteristics for all components); d) materials processing methods needed for a large-scale manufacturing; e) method of the application of the material to the structure; f) assurance of confidence in the performance of the system throughout the mission sequence environments.

The performance of the heat shield and its response to the environment depends not only on the basic properties of the material itself but also on the environment it is exposed to. While it is relatively easy to predict analytically the effect of the substructure on the heat shield material and verify it during the ground test program, it is extremely difficult to predict the heat shield performance for a particular application without an extensive testing program. There are no ground test facilities available now or projected in the near future capable of simultaneously duplicating or simulating the anticipated flight environment parameters. Such simulation, of course, would be necessary to assure the conformance of the preflight prediction with actual flight data for a material which was not flown before. The necessity of flight testing (assuming the existence of an extensive ground test program) depends on the degree of the conformance required of the design, which in turn depends on the safety margins allowed. It is not possible to design a heat shield with any degree of confidence without an extensive material characterization program including more than just "simulated" quasi-steady state entry heating arc-jet tests. The possibility of transient trajectory simulation in the arcs greatly enhances the predictability.

The mission system and subsystem specifications (Volume III, Book 2 and Volume V, Book 2) define the requirements imposed on the thermal protection system by the design selected. The thermal protection must survive the decontamination and sterilization environments, mechanical environments, possible exposure to vacuum, low temperatures anticipated in space, and then perform its thermal function in the entry environment. A summary of the thermal protection system elements is shown in Table VII, which indicates the critical items and the general purpose of the tests required to assure performance during specific phases of the mission.

2.2.2 Test Objectives and Description

2.2.2.1 Heat Shield Material Characterization

The analysis of the conceptual design of the Mars Probe (EFO) contained in the other volumes of this report indicates that the development problems usually encountered in the entry vehicle technology exist in this application as well, and therefore can be handled by the existing techniques and facilities. However, this technology does not allow the desired degree of simulation of flight parameters during the ground testing phase, since the simulation of transient heating, enthalpy, pressure and heat pulse is difficult to obtain. Thus only [illegible] characterization of materials is feasible in ground tests. The lead times associated with the transformation of a laboratory material into a sufficiently characterized and manufacturable item are critical. Also, the influence of (a) the environments associated with the decontamination and sterilization procedures; (b) long time exposure to space vacuum; and (c) the effect of Martian atmosphere(s) must be considered in the material selection.

A small body of the material properties and characteristics is now available having been developed concurrently with the present program. Basic understanding of the ablation mechanism for the materials under consideration has been achieved for the air atmosphere, and the effect on thermal and mechanical properties of ETO decontamination and the sterilization dry-heat cycle has been partially determined. The latter is mandatory to predict material performance. Thus, for conceptual design purposes, it was possible to design and evaluate the requirements for the thermal protection system. However, such information is not sufficient for design of hardware to the

TABLE VII

THERMAL PROTECTION TEST SYNTHESIS SUMMARY

Item No.	Primary Operating Function of the Elements(s)	Critical Area	Description of Elements Subassembly, etc.	Phase of Mission of Concern	Design Information	Performance Prediction	Flight Test Design Verification Required	Mfg. Concepts and Methods Test
1	Thermal Protection	Yes (characterization)	Ablator	Entry	X	X	Yes*	X
2	Thermal Control and Structural	Yes (for use of properly conditioned material properties)	Ablator	Spaceflight	X	-		
3	Thermal Control and Structural		Ablator	Decontamination/ Sterilization	X	-		
4	Thermo-structural	Yes (for manufacturing)	Bond	Assembly to parachute deployment	X	X	Yes*	X
5	Thermal Protection	Yes (overall performance verification)	Ablator/Bond/Structure	Entry	-	-	Yes*	X
6	Thermo-Structural Compatibility	Yes (mechanical integrity)	Ablator/Bond/Structure	Decontamination/ Sterilization		(See Structures Test Program)		
7	Thermo-Structural Compatibility	Yes (mechanical integrity)	Ablator/Bond/Structure	Decontamination/ Sterilization				
8	Thermal Control	Yes (for thermal control function)	Heat Shield/Thermal Composite	Decontamination/ Sterilization/spaceflight	See Thermal Control Development Program			See Thermal Control Development Program
9	Thermal Protection	Yes (for aggravation areas).	Joints, interfaces, protuberances, etc.	Post Separation/ Entry	X	X	Yes*	
10	Thermal Control	Yes (for thermal control function)	Joints, interface, protuberances, etc.	Spaceflight Decontamination/ Sterilization	See Thermal Control Development Program			
11	Thermal Control	Yes (for thermal control function)	Joints, interfaces, protuberances, etc.		See Thermal Control Development Program			

*Note: depends on the feasibility of trajectory simulation in the arcs.

required degree of reliability since the calculated heat shield weight fractions appear to be relatively high precluding the use of large safety factors.

1. <u>Design Information Acquisition for Entry Phase</u> -- The purpose of the heat shield material test program is to determine the thermal, optical and mechanical properties, and ablation characteristics of existing materials for design use (determination of heat shield thickness required) rather than development of new materials.

The program should consist of preliminary screening tests and subsequent comprehensive development tests. No more than four materials should be used for the screening tests and no more than two materials should be considered for the development tests: one for the reference design and one for backup. The Purple Blend, Mod 5 and Cork Silicone are the most likely candidates as of now. Purple Blend was used as the reference in the conceptual design studies.

The scope of the screening tests in terms of individual objectives, test conditions and their range, number of tests and samples, test procedures and techniques is outlined in Table VIII. It indicates the number of tests at various points in the desired range for various conditions of the specimen prior to test. The number of tests presented is for the purpose of comparison with the development test program requirements. The table describes the type of test (including the candidate facility, where pertinent) to be performed to obtain the necessary screening data. Two sets of candidate materials and samples would be expos [illegible] sterilization cycle first to determine the effect of this environment, then one of the set of the samples would be exposed to the space vacuum simulation and tests would be repeated (see Section below).

After completion of the screening tests, the selected material(s) would be evaluated in a more comprehensive characterization program as described in Table IX. This program involves the same and additional tests as included in the screening tests and will completely characterize the remaining candidate material(s) to allow final choice of a material.

2. <u>Design Information Acquisition for Sterilization and Spaceflight Phase</u> -- Information obtained for the entry evaluation will be used for this phase of the development specifically and is described here in more detail.

a. Mechanical and Thermal Properties -- The mechanical properties of the candidate heat shield materials should be determined after exposure to ETO decontamination, dry-heat sterilization and exposure to the vacuum and temperature conditions of outer space.

TABLE VIII

HEAT SHIELD MATERIALS SCREENING TEST

(PRECONDITIONED) DECONTAMINATED, STERILIZED AND EXPOSED TO SIMULATED SPACEFLIGHT

Element	Mission Phase of Concern	Design Analysis Problem Areas or Requirements	Test Objectives	Test Description	Test Conditions Desired
Ablator	Entry	Selection of efficient lightweight material and preliminary design for the expected thermal environment (atmosphere heat flux and duration, enthalpy and pressure)	1. Provide basic characterization of materials for design calculations of temp., mass loss, required thickness, leading to selection of material(s) for min. weight fraction (performance prediction).		1. Number of materials not to exceed 4. 2. Environmenta. test parameters or their derivatives to approach the design operating conditions. 3. No. of tests will depend on reliability requirements.
			a) Determine thermal properties	Measurement of thermal conductivity	True virgin materials and three fully charred samples Temp. range -50°F to surface temperature expected
				Measurement of heat capacity	Same as above but 2 samples only
			b) Determine optical properties	Measurement of thermal emittance	Same as heat capacity
				Measurement of transmittance/reflectance	Same as above
			c) Determine other chemical and physical properties	Measurement of density	Same as conductivity
				Measurement of porosity	None for screening
				Measurement of permeability	None for screening
				Measurement of internal rate constants	3 temperature rates
			d) Determine ablation characteristics and flow effects	Measurement of laminar ablation parameters	Five samples H_m/RT_o:50-200; q_c as required
				Measurement of turbulent ablation parameters	
			2. Verify theoretical ablation model usage of degradation parameters, surface and internal reactions, blowing and atmosphere	Measurement of ablation rates, weight loss, density changes and temperature distribution under simulated entry conditions for a thermocouple instrumented sample transient test.	13 samples H_m/RT_o:50-200; $T_{structure}$ as required by design (Approx. 500°F at the bond line) Atmospheres: air and 2 other compositions.
			3. Provide preliminary design information on mechanical behavior of materials to assure integrity and compatibility with the structure.		
			a) Determine tensile properties	Experimental determination of stress-strain curves and measurement of the thermal strain	Five samples of each test Temperature range - 150 to approximately 500°F.
			b) Determine compressive properties	Experimental determination of stress-strain curves and measurement of the thermal strain	Five samples of each test Temperature range - 150 to approximately 500°F.
			c) Determine Poisson's Ratio	Experimental determination of stress-strain curves and measurement of the thermal strain	Five samples of each test Temperature range - 150 to approximately 500°F.
			d) Determine Thermal Expansion	Experimental determination of stress-strain curves and measurement of the thermal strain	Five samples of each test Temperature range - 150 to approximately 500°F
	Decontamination/ Sterilization Spaceflight	Changes in material decomposition and behavior during these phases of mission and the ensuing difficulties in cost control, material selection, evaluation, design and test	1. Select material requiring minimum preconditioning treatment needed to minimize changes due to the decontamination/and sterilizing cycles and vacuum exposure.	Measurement of selected thermal properties and ablative characteristics.	a) same as for entry but material decontaminated and sterilized only b) Same but also exposed to simulated space condition.
			2. Adjust composition to minimize degradation and provide maximum stability	Measurement of mechanical properties	a) Same as (a) above b) Same as (b) above
				Determination of chemical composition by infrared spectrophotometric and gas chromatography studies.	a) Material decontaminated and sterilized, process simulated b) As decontaminated, sterilized c) As exposed to decontamination, sterilization and space conditions
Bond	All Phases	Bond strength at elevated temperatures	Provide thermal properties for design.	No thermal screening required. Manufacturer's data to be used in preliminary design. (See also Structures Testing).	

TABLE IX

HEAT SHIELD ABLATOR DEVELOPMENT TEST
(Sterilized and preconditioned)*

Element	Mission Phase of Concern	Design Analysis Problem Area or Requirements	Test Objectives	Test Description	Test Conditions Desired
Ablator	Entry	1. Determination of H/S weight fraction and prediction of response of the H/S to expected environments.	1. Provide property & characteristics parameters for design use & performance prediction.		1. No. of materials not to exceed 2. 2. Environmental test parameters of their derivatives to approach the design operating conditions. 3. No. of tests will depend on reliability requirements.
		2. Preparation of H/S Material specifications	a) Determine thermal properties	Measurement of thermal conductivity	Six virgin mat'ls and six fully charred samples. Temperature range -50° to surface temperature expected.
				Measurement of heat capacity	Same as above but 4 samples only
			b) Determine optical properties	Measurement of thermal emittance	Same as heat capacity
				Measurement of transmittance/ reflectance	Same as above
			c) Determine other chemical & physical properties	Measurement of density	Same as conductivity
				Measurement of porosity	To be determined after screening tests
				Measurement of permeability	Same as above
				Measurement of internal rate constants	Three temperature rates
			d) Determine ablation characteristics and flow effects	Measurement of laminar ablation parameters.	10 samples H_m/RT_o 50 -200; q_c and p as required.
				Measurement of turbulent ablation parameters. subsonic	Six samples H_m/RT_o 50-200; q_c and p as required
			2. Verify theoretical ablation model usage of degradation-parameters, surface & internal reactions, blowing and atmosphere.	Measurement of ablation rates, weight loss, density changes and temperature distributions under simulated entry conditions for a thermocouple instrumented sample transient test.	25 samples $\bar{H}_m/RT_o$ 50-200; q_c and p as required T (structure-as required by design (approx. 500°F at the bond line) Atmosphere: air and 2 other compositions.
		3. Compatibility with the structure.	3. Provide design information on mechanical behavior of materials to assure integrity & compatibility with the structure.		
			a) Determine tensile properties b) Determine compressive properties c) Determine Poisson's Ratio d) Determine thermal expansion	Experimental determination of stress-strain curves and measurement of thermal strain	Five samples for each test Temperature range-150 to approximately 500°F.
	Decontamination/ Sterilization/ Spaceflight	Effect of Heat shield exposure to decontamination, sterilization and space vacuum on its performance	Provide design information for the material performance evaluation in "as exposed" condition for thermo-structural & thermal control performance prediction.	See tests for entry phase, and screening program.	
	Post Separation	Impingement of ΔV rocket plume on heat shield.	Determine plume heating and its effect on heat shield performance.	Exposure to rocket plume	Actual motor in Vacuum
	Misc. Environments	Assurance of performance and satisfaction of specifications.	As required by Government specifications		
	Manufacturing	Assurance of reproducibility of materials, homogenity and integrity during exposure to various elements.	1. Raw materials a) Identify and control contamination. b) Determine batch to batch chemical variation. c) Control moisture.	See description in Section	
			2. Develop process for scale-up from laboratory techniques and select fabrication process.	Depends on the screening test results and selection of reference materials.	
			3. Develop nondestructive test method.		
			4. Verify heat shield process (including humidity effects)		
Bond	All phases	Same as for the ablator except no requirement for ablative performance test. See Structural Test.			
Thermal Control Coating/ H/S/Bond	All phases	No critical thermal protection problem areas.	1. Determine thermostructural compatibility.	See Structure Test	
			2. Determine thermal control/ heat shield material compatibility	See Thermal Control Test.	
Structure Composite					
Joints, Interfaces, Protuberances	Entry	1. Prediction of H/S performance in areas where potential aggravation problems exist. (See also Aerothermodynamic tests - difficulty in environment prediction.)	1. Determine empirically the effect of aggravation on material response	Measurement in a joint test with aerothermo changes in the environment. Measure the erosion (ablation) rates in the vicinity of the aggravation, together with temperature response.	Depends on design configuration
		2. Selection of local substitute materials or design configurations to assure performance.	2. Predict Performance.		

*Unless otherwise noted

Two sets of test specimens of each material should be fabricated and subjected to the conditions of decontamination and dry-heat sterilization as specified in JPL specifications XS-30275-TST-A. After exposure to decontamination and dry-heat sterilization, one set of test specimens should be utilized to determine the effects of this environment. The second group of samples would be exposed to the vacuum and temperature conditions of space to determine the effects of the combined environments on the mechanical properties.

At the completion of each environmental exposure the following mechanical properties should be evaluated and the effect of exposure determined by comparing the results with properties obtained on control specimens from the same batch of material: 1) tensile properties over a temperature range of -100 to 350°F; 2) compressive properties over a temperature range of -100 to 350°F; 3) shear properties over a temperature range of -100 to 350°F; 4) dimensional stability, i.e., weight loss and dimensional changes of test specimen; 5) hardness, shore A or equivalent.

b. Chemical Composition -- The chemical composition of the candidate heat shield material and their degradation products should be analyzed after dry-heat sterilization and vacuum exposure to identify degradation products and assess their affects on 1) the thermal control coating; 2) antenna windows; 3) instrumentation and 4) control mechanism within the canister, and to guide formulation studies to adjust ablator chemical composition to minimize degradation and provide maximum stability.

1) Decontamination and Sterilization Effects -- The changes in ablator chemical composition due to above conditions should be determined as follows:

a) Infrared Spectrophotometric Studies - Samples of the decontaminated and not decontaminated heat shield material to be evaluated will be enclosed in infrared gas cells. The cells will be evacuated several times and filled to a slight positive pressure with an inert gas such as nitrogen or argon. The entire cell containing the sample of heat shield will be heated in a laboratory oven at 135°C for 36 hours. At the end of this period, the cell will be placed in the beam of the infrared spectrophotometer, maintained at 135°C by means of heating tapes to prevent any condensation, and the chemical composition of the evolved gases determined by a complete infrared scan.

Once the evolved gases have been identified, a quantitative analysis of the gas mixture will be conducted by infrared spectrophotometry. Calibration curves will be prepared for each of the components in the gas mixture using a vacuum sampling apparatus recently constructed for the determination of HCl in BCl_3.

b) Gas Chromatography Studies -- The results of the infrared studies will be confirmed and amplified by gas chromatography analyses. Once a qualitative identification of the gases evolved during exposure has been made, the proper columns for separation and the proper conditions for quantitative determination of the gases will be chosen. Gas chromatography will not only confirm the infrared work but will provide an analysis for any gases (such as hydrogen or oxygen) which do not have infrared absorption bands and are thus not detected by infrared spectrophotometry. Gas chromatography will also provide a more quantitative analysis on the total gas mixture evolved from a heat shield material than will infrared spectrophotometry.

For each material investigated the preliminary study will be conducted using both infrared and gas chromatographic analyses in a complementary manner. Subsequent, repetitive studies on the same heat shield material will be conducted by gas chromatography once the proper column conditions have been established.

c) Decontaminated and Sterilized Heat Shield Composition - The studies described in sections a) and b) above will define the gases evolved from a heat shield material during sterilization (with and without exposure to ETO). A companion study will be made to determine the chemical changes in the sterilized heat shield.

It is entirely possible that the upper portion of the heat shield will evolve and lose as gaseous products more material than the lower portion of the heat shield. Therefore, a chemical analysis profile will be conducted on each type of heat shield material which is subjected to sterilization. Layers of the heat shield material will be machined off by precise techniques already in use for the analysis of charred composite materials. The individual layers will be analyzed for resin and ash content. On the basis of these analyses, selected layers will be examined by infrared spectrophotometry to determine changes in the chemical structure of the polymeric component of the heat shield. In this manner a profile of the changes induced in the heat shield by sterilization will be constructed.

2) Vacuum Exposure Effects -- Samples of material which have been subjected to the decontamination and sterilization cycle will be tested for any possible effect of long term exposure to the vacuum and temperature conditions of outer space. The material

will be suspended from the beam of a Cahn balance down into a hangdown tube. The hangdown tube will then be cooled by an appropriate bath. The Cahn balance is enclosed in a glass pig and integrally connected to the hangdown tube so that the entire apparatus will be evacuated to 1×10^{-6} torr. Then the change in weight with time will be followed automatically and with high sensitivity since the Cahn balance will detect weight changes on the order of 1 microgram.

If a loss in weight is indicated by the vacuum exposure simulation tests, the exposed material will be analyzed by the same layering and chemical analysis techniques discussed above to determine the nature of the chemical changes induced in the heat shield material.

3. Performance Prediction Testing for Entry Phase -- All information required for design use is required for performance prediction. However, the ablation tests in the OVERS or similar facility are conducted primarily to verify the applicability of analytical methods combined with the use of independently measured properties.

a. Nondestructive Test Development -- Several immediately recognizable nondestructive problem areas exist for which tests must be provided to assure reliability of the design.

Experience has shown that for the heat shield the existing nondestructive test approaches and conventional equipment are incapable of assuring compliance with design requirements of low density or elastomeric materials. Problems associated with the inspection of these materials such as low specific gravity which results in high acoustic attenuation and the possibility of direct application of the heat shield to the substructure which will require one sided radiometric density determinations are not considered insurmountable, however, development of proper nondestructive test techniques will require new approaches and facilities.

b. Heat Shield Process Verification -- The process for fabricating the heat shield should be verified by destructively testing three heat shield shapes. The ablator will be fabricated as specified in the preliminary process specification, released at the completion of the fabrication evaluation phase of the development program. The shape of the sections produced will be similar to the final design but may have to be increased in size to provide sufficient material to conduct the required tests. The data generated will be analyzed to determine the reproducibility of the fabrication process and to confirm design properties.

The following properties will be determined: 1) tensile properties, 2) compressive properties, 3) shear properties, 4) ablative characteristics, 5) thermal conductivity, 6) specific gravity, 7) thermal behavior (TGA), and 8) porosity.

2.2.2.2 Bond Performance and Properties

1. <u>Thermo-structural Design Information Acquisition and Performance Prediction Testing</u> -- As in the case of the heat shield material, no new bonding agents will be developed but existing materials will be selected. No critical problems specific to the probe development are expected to arise; however, a proper bonding of the heat shield material to the structure is of utmost importance in the fabrication of the shell. The information necessary for design includes the mechanical and thermal properties especially the bond shear and tensile strength, thermal conductivity, heat capacity, and temperature limitations. These will have to be obtained in the course of the program as they are needed for design and evaluation of performance during all phases of probe development and flight.

The objectives of this program are to obtain one or more suitable adhesives for bonding the various heat shield materials to the substructure. A wide range of environmental conditions will be encountered during storage and in flight. Bond-line temperatures may be as low as -100°F. Vibration, acoustic noise, shock, acceleration, sterilization and vacuum exposure are other environmental conditions that will be improved during the life of the probe. The adhesive that bonds the heat shield to the substructure of the vehicle has to withstand thermal and mechanical stresses with a maximum degree of reliability.

The program recommended to select an adhesive will include: 1) Comparing various classes of structural adhesives by means of shear moduli determinations at several temperatures. The classes of adhesives include epoxies, modified epoxies, silicones and other elastomers. 2) Selection of one or more classes of adhesives will be made from a comparison of shear modulus and shear stress versus design allowables if feasible. A more intensive evaluation of physical properties and determination of the resistance to the effect of environmental and simulated flight conditions may also be required. Representative adhesives would be tested both as a component in a composite structure and separately as a material to determine strength values in tension, compression and shear as a function of temperature, and time at temperature. Resistance of the composite to thermal shock sterilization and vacuum exposure will be measured.

2. Manufacturing Concepts and Methods Test

a. Bonding Process and Surface Preparation -- Assurance of the integrity of the heat shield bonding to the structure often presents critical fabrication problems. For example, one of the concepts of attachment of the Purple Blend Mod 5 ablator to the forebody structure anticipates use of fiberglass ply with loops extending into the ablator to improve the strength of the activated (charred) material as a part of the bonding concept. Such a concept may require a great deal of development work to assure performance of the full-scale article. The anticipated test program would include the following efforts:

1) Methods will be explored for improving existing adhesives of the selected class or classes if required. The effects of loops, fillers, and fibers on physical properties will be determined with the objective of improving adhesive strength and reliability.

2) The several factors involved in the bonding process, including methods of application and curing conditions, will be investigated for those adhesives that have been selected for intensive evaluation. Procedures that are most readily applicable to production will be emphasized, but the physical and thermal properties of the final structure in terms of design requirements will be the most important criteria for the final selection.

3) General methods of surface cleaning of substructure surfaces will be evaluated concurrently. These methods include: chemical cleaning-non etch; chemical cleaning with surface etching.

The criteria to be used for the selection of a method of surface preparation will be attainment of reliable bonds of adhesive to adherents that meet design requirements as well as a process that is feasible for full-scale production.

4) The heat shield materials under consideration are so dissimilar that primer systems will be of great importance. In the case of glass-reinforced silicone cork, surface treatment will be necessary. There is very little information available to date on the bonding of glass reinforced silicone cork. Were this type of material selected for use, a development effort will be required to attain suitable cleaning methods.

b. Nondestructive Test Development -- As noted the requirement of a reliable heat shield-to-structure bond is often of vital importance to the performance of the flight article. Thus it requires a reliable method of assurance of bond integrity to avoid unbonded areas. For structures of the anticipated size the problem may be critical.

The presence of the low density or elastomeric heat shield material results in a large acoustic impedance mismatch and poor ultrasonic and infrared response to unbonding and presents a unique bond inspection problem. Again, the problem requires effort beyond existing technology.

2.2.2.3 Ablator/Bond/Structure Composites' Performance

Ablator/bond and ablator/bond/structure composites are certainly the important building blocks of the heat shield, since they perform the basic thermal protection function during entry; however, they do not present (in composite form) any critical thermal heat shield development problems. The tests required to verify the thermostructural performance prediction are described in Section 2.3 (Structures Development).

2.2.2.4 Heat Shield/Thermal Control Coating Composites

The presence of the thermal control coatings on the heat shield is not likely to affect its primary function during the entry. Therefore no development program is required from the heat shield design point of view. The inverse problem does, however, exist. The development program of thermal control system (Section 2.4) describes the necessary effort. Also the development work in the area of the heat shield material fabrication and formulation includes effort directed toward minimizing the detrimental effects of ablator outgassing during critical phases of mission (sterilization and flight in space vacuum).

2.2.2.5 Joints, Interfaces, Protuberances Behavior and Miscellaneous Thermal Development Tests

As the detailed design of the flight capsule system progresses and manufacturing assembly procedures are established, the ideally projected, smooth and uninterrupted heat shield surface is perturbed. Various subsystems or components are located in the vicinity of the heat shield disturbing the flow field and locally increasing the heat inputs; manufacturing joints, access or attachment points for other parts of the system may create local ablator discontinuities, cavities, flats or protuberances; provision of a separable nosecap to facilitate TV operation may create local disturbance of heat flow pattern; or operation of the ΔV rocket may result in additional heating from the plume.

Some of these problems may be solved analytically, but most will require separate developmental tests either to provide direct experimental information for the thermal design or to establish the flow field in the vicinity of aggravations. Some of the projected tests are indicated in Tables VIII and IX. Others will have to be tailored to the actual geometry of the anticipated source of the problem. The aero-thermodynamic tests leading to the determination of the flow field and heating in the vicinity of protuberances were described previously. The criticality of the problem is best illustrated by some of the similar tests indicating increases in local heating of an order of magnitude. In such cases, local heat shield material would have to be changed and higher density inserts provided. This in turn would affect manufacturing methods and create secondary discontinuities.

Arc tests as well as testing in rocket engine exhausts to determine aggravations are recommended.

1. Protuberances -- A thermo-structural design evaluation test is envisaged to ensure the design adequacy of the capsule heat shield and structure in the vicinity of the attitude control nozzles. The outboard nozzle bow-shock and body boundary-layer interaction will create increased local heating and heat shield degradation.

The test program may utilize the rocket engine exhaust facility at the Malta test station in upstate New York. This facility, while not simulating the reentry gas flow chemistry, can accommodate a model up to 12 inches in diameter and subject it to the following maximum conditions:

Heat Flux	400-900 BTU/lb
Enthalpy	2450 BTU/lb
Shear Stress	10-20 lb/ft^2
Stagnation Pressure	10 atmospheres
Total Temperature	5940°R
Test Duration	200 seconds

Similar flow-field interaction programs have been conducted in this facility on finned bodies as well as other types of protuberances.

The proposed test program will utilize an 10 x 10-inch full-scale section of the heat shield, structure and outboard attitude control nozzle cluster and subject this prototype model to the rocket exhaust environment which most nearly simulates the peak entry flight conditions. Data acquisition for this test will be completely optical, photographic and pyrometric, together with the appropriate pre- and post-test dimensional change data.

2. Surface Gaps and Interfaces -- The general concern is for possible damage to the heat shield and backup structure due to accelerated erosion and heating at a cavity. The cavity may originate from an interface or it may be a break in the heat shield for such items as the ACS cold-gas jets. A surface cavity degrades heat shield performance in two unpredictable ways:

a. Erosion -- at the cavity may be accelerated due to increased local shear and heating on a structurally discontinuous ablative surface.

b. Heat Transfer -- at the bottom of a cavity may be accelerated, thus exposing the backface to intolerable temperatures.

The complexities of the interrelated effects on the flow field, heat transfer rates, and cavity erosion prevent their direct solution in a mathematical model for heat shield design. To assure a successful design the gaps such as the one between the nosecap and main body of the vehicle may be tested using a scale model in the Malta test facility.

Data acquisition for this test will be the same as for the protuberance tests described in the previous section.

2.2.3 Test Facilities, and Equipment

It was noted in the preceding sections that the critical problem to be encountered in this development program was not so much the availability of the technology to perform the testing but rather the time element required to scale the material up from the laboratory status to a full-scale manufacturable product, and to actually test the final product. It is thus of essence to have readily available test facilities and equipment and established techniques to perform the necessary tests. Such facilities, equipment and operating procedures as are required are described in the following sections.

2.2.3.1 Thermal and Optical Properties Testing

A summary of equipment and its operating units for various thermal tests is given in Table X.

TABLE X

THERMAL PROPERTY DETERMINATION EQUIPMENT AND TEMPERATURE RANGE

Property	Equipment	Equipment Temp Range
Thermal	Guarded Hot Plate	-250 to 1000 °F
Conductivity	Radial Flow	500 to 5000°F
Enthalpy	Method of Mixtures	-320 to + 1800°F
Specific Heat	Differential Scan Calorimeter	-150 to + 900°F
	Method of Mixtures	-320 to + 1800°F
	Bunsen ICC Calorimeter	-320 to + 2500°F
	Pulse Technique	+1500 to + 5000°F
Emissivity	Barnes Radiometer	to + 5000°F
	Copper Sphere	-200 to + 3000°F

Automation of the guarded hot plate apparatus would offer the capability of measuring 14 tests per unit used (each test consists of two specimens and five temperature levels) in 1 week. Automation of the specific heat apparatus would provide a capacity of 35 tests (each consists of five specimens, 16 measurements of enthalpy change from five different temperature levels in 1 week. Automation of both of these tests would significantly reduce the manhour requirements that are needed for non-automated systems.

2.2.3.2 Chemical and Other Physical Properties Testing

1. Density -- A Beckman Air Comparison Pycnometer, Model 930, could be used for the measurement of the volume of solid samples. A modification of the pycnometer to permit purging of the sample by vacuum pumping and measurement in an inert atmosphere of nitrogen or helium permits the measurement of true volume of active materials.

The sample size must be such as to fit in the cylindrical holder which is 1.5 inches in diameter by 1.5 inches in height. If possible, solid materials should be geometrical in shape so that the bulk volume may be determined by micrometer measurements. If the sample is not geometrical, the bulk volume may be determined, after measurement of apparent volume, by immersing the sample in molten paraffin. The temperature of the sample is allowed to come to equilibrium with the temperature of the wax, then removed and cooled to room temperature. Any excess wax should be removed by shaking the sample while the wax is still molten. This process impregnates the open pores with solid wax which allows measurement of the bulk volume. The weight of the sample is taken before the wax treatment.

Measurements of bulk density by weight and volume measurement are very precise. Duplicate samples are sufficient.

2. Porosity -- Various means of measuring porosity are used. An apparent porosity may be calculated from the apparent and bulk volume. Likewise the total porosity may be determined from the bulk density (BD) and the true density (TD). The true volume of the material (excluding the volume of open and closed pores), is determined by grinding the sample to fine particles and measuring the volume occupied by the particles in the Beckman air comparison pycnometer. The percent true or total porosity may then be calculated as 100 (TD-BD)/TD.

For pore-size and pore-volume distribution the Amino-Winslow Porosimeter [ASTM Bull., No. 236, 39 (1959)] could be used.

3. Permeability -- The gas permeability of polymer is a basic property of the material independent of specimen geometry. It is related to the diffusion rate and solubility of a gas in a material by the equation $\overline{P} = DS$ where $\overline{P}$ = gas permeability, D = diffusion rate and S = solubility. The gas permeability is normally assigned a value identical to the gas transmission rate of a specimen of unit thickness.

Gas transmission rates should be measured by ASTM Method D1434-63 Gas Transmission Rate of Plastic Film and Sheeting. The gas transmission rate is the steady-state volume of test gas that passes through a known area of a specimen of known thickness per unit of time. It must always be related to specimen thickness and test temperature. Thus, if specimens of various materials are measured in the form of sheets of equal thickness at one temperature (room), a relative comparison of gas transmission may be made. For gas transmission rates a minimum of three specimens should be tested at each condition of test material, gas and pressure.

If absolute gas permeability is desired, the rate of gas transmission and the diffusion constant will be determined by the time lag technique. The apparatus designed by Yasuda and Stannett [J. Polymer Sci. 57, 907--923 (1962)] , should be used.

4. Thermal Gravimetric Analysis (TGA) and Differential Thermal Analysis (DTA) -- Small powered samples are suitable for both TGA and DTA. Both methods are very reproducible so that two or three curves are sufficient to characterize a material in any one atmosphere.

TGA-DTA thermoanalysis equipment manufactured by the Harrop Precision Furnace Company could be used for TGA to 1600°F and DTA to 1900°F. A TGA apparatus assembled from a Lahn microbalance, a Marshall Pt-Ir wound furnace, and F&M temperature programmer and a Houston X-Y recorder could be used for work to 2800°F with milligram size samples.

2.2.3.3 Ablative Characteristics Testing

Table XI presents a summary of the operating characteristics of the Avco arc facilities. It should be noted that the tests could be conducted in facilities other than Avco's if available and if they provide better simulation of entry parameters.

1. Model 500 Arc -- The Avco Model 500 plasma generator is generally used in support of screening test programs and for obtaining fundamental ablation data. The gas environment most used consists of a subsonic jet (1/4 to 1 inch in diameter) utilizing air as the working fluid (although gases such as N_2, helium, CO_2, O_2, argon, or mixtures thereof have also been used). The sample can be either flat face, hemispherical, or conical, and can be instrumented. Gas enthalpies and heat-transfer rates (flat-faced cylinders) that can be generated cover the range of 600 to 10,000 Btu/lb and 25 to 1300 Btu/ft^2-sec, respectively.

2. OVERS Arc -- The OVERS facility consists of an electric-arc gas heater with a 3-inch diameter exit nozzle, and a 500-kw rectifier and is connected to a 33,000 ft^3/m (at 1×10^{-1} torr) central vacuum system through a 24-inch throttling valve. Arc operation is in nitrogen with 23 percent oxygen injection in the arc plenum . OVERS is capable of operation at enthalpies up to 26,500 Btu/lb and at pressures in the range of 0.01 atmosphere.

The usual test technique that is used for the simulation of heating and environment flow conditions is that of the stagnation or splash model. The splash test is arranged such that the heated air exits from the supersonic nozzle to the ambient (vacuum) surroundings; the sample is externally mounted and swung into the stream. The sample size can be varied from 1 to 4 inches in diameter and the sample shape is usually a flat-faced cylinder.

TABLE XI

SUMMARY OF AVCO ARC SIMULATION PARAMETERS ON STAGNATION OR SPLASH MODELS
(Actual Test Conditions Achieved)

Simulation Parameter	Orbital Vehicle Reentry Simulator OVERS Arc	Radiation Orbital Vehicle Reentry Simulator ROVERS Arc		10-Megawatt Arc	Model 500 Arc
		Convection*	Radiation**	Open Jet	
Sample Diameter inches	4	4	1	2	3/4
Enthalpy (H/RT_o)	100, 300, 600, 800	100, 300, 600, 800		50, 100, 200, 250	100, 200, 300
Cold-Wall Stagnation Point Heat Flux Btu/ft^2sec	15, 150, 300, 700	15, 150, 300, 700	700^2	300, 1100, 3000, 4000	800, 1300, 1300
Nozzle Diameter (inches)	3	3, 5	--	0.9, 1.25	1/4 to 3/4
Plenum Pressure	15 to 100 mm Hg	15 to 760 mm Hg	--	2 to 20 atm	1 to 2 atm
Gas	O_2, N_2, Argon, CO_2, Helium	O_2, N_2, Argon, CO_2, Helium	--	air or other gas mixtures	O_2, N_2, Argon, Helium Mixtures
Test Duration	Continuous	Continuous	Continuous	60 seconds	60-600 seconds
Jet Diameter, inches	3	6	--	2	1/4 to 1

*Using the presently operational OVERS arc head
**Presently operational radiation lamp operating at moderate power levels.

a. Trajectory Simulation -- An attempt should be made to subject samples to simulated flight trajectories. Inasmuch as the OVERS arc can be utilized in such a continuous test by changing mass flows and power, it is suggested that such a series of tests be run on those materials which appeared promising in the previous steady-state tests. When changing test conditions, approximately 20 to 40 seconds are required to make the necessary adjustments in gas mass flow and power.

b. Detailed Design Simulation -- The effects of bond lines filled joints, protuberances, gaps, etc., on local material performance should be tested.

3. 10-Mw Arc Facility -- The basic components of the arc include a 4-inch-diameter spherical plenum chamber into which four individual arc heads exhaust radially. The four arcs are mounted in a common plane and are equally spaced at angles of 90 degrees around the periphery of the plenum chamber. The heated air mixes in the plenum chamber and exhausts through an exit nozzle in a direction perpendicular to the plane of the four radial plasma generators.

The power supply for the unit is a group of 2080-12 volt heavy-duty truck storage batteries. The arcs are initiated by means of fine tungsten wires. Air is injected tangentially into the arc chambers through sonic orifices and flows out of the exit nozzle after passing into the plenum chamber. When the power breaker is closed, steady-state values of the current, voltage, and plenum pressure are achieved in less than 1 second.

The 10-Mw arc is a flexible test facility which can be utilized in several configurations. The simplest of these is the splash test in which the material specimen is placed in the laboratory atmosphere directly in the exhaust jet from the plenum chamber. This configuration produces laminar, stagnation-region flow over a flat-faced specimen.

The subsonic pipe test is used to obtain turbulent heat of ablation data on ablative materials. In this type of experiment, a pipe of the material to be tested is mounted in the facility between the plenum chamber and the sonic exit nozzle. The high-enthalpy air in the plenum chamber flows through the specimen and exhausts into the atmosphere after passing through the water-cooled sonic nozzle downstream of the specimen. The standard sample configuration employed in this type of experiment has inside and outside diameters of 1.25 and 3.0 inches, respectively. The overall specimen length is 5.0 inches.

In performing a subsonic turbulent pipe test in the 10-Mw facility, it is not possible to make observations of the ablating surface during the experiment. Hence, certain input parameters (surface temperature, emissivity, surface radiation, and the time at which ablation starts) must be arrived at by use of experimental measurements obtained in other arc facilities (Model 500 arc) or computational procedures. Flow conditions within the pipe sample are assumed to be such that a turbulent boundary layer exists rather than fully developed turbulent pipe flow.

In the sonic nozzle test procedure, the water-cooled sonic exit nozzle usually employed is replaced by a nozzle fabricated from the material to be tested. The nozzles used in this type of test have an initial throat diameter of 0.50 to 1.2 inches. Since the throat diameter increases with time due to material ablation, and the air mass flow through the sample is maintained at a constant value, arc-plenum pressure decreases through a major portion of the experiment. In general, arc efficiency tends to increase with decreasing pressure; hence gas enthalpy increases throughout the test. As a consequence of these variations, the sonic pipe test can be considered to be transient in nature. This is in marked contrast to the quasi-steady experiments performed with the subsonic pipe test technique.

4. ROVERS Arc Facility -- The convective splash tests using a 5-inch nozzle exit diameter and tests requiring a model stagnation pressure of 0.1 atmosphere could be carried out in the facility referred to as the radiation orbital vehicle reentry simulator (ROVERS).

This facility is currently in the final checkout stages of its construction. The combined convective and radiative facility, utilizes four radiation sources together with a family of convective arc sources depending upon the desired jet enthalpy and pressure level. The double-walled, water-cooled, test tank is 6 feet in diameter and 16 feet long. This tank will be able to accommodate samples 6 to 12 inches in diameter. A probe table is being constructed and will be available to handle the various diagnostic probes (pressure, enthalpy, and heat flux) as well as sample models. Several viewing ports are present to allow for additional instrumentation to study material behavior.

Convective heating simulation is provided by a centrally located arc heater such as used in the OVERS arc. In addition a 1500-kw high enthalpy arc heater is currently being developed to expand the test range to higher pressures (up to 3 atmospheres) at comparable enthalpies. For convective splash tests, the ROVERS will be operated in a manner very similar to that of the OVERS.

The radiant sources that are being used in the ROVERS facility are lamps using a vortex-stabilized arc concept. A separately excited magnetic field diffuses the discharge in the anode region.

The ROVERS arc utilizes (at present) the operational arc head used on the OVERS facility. As part of the continuing Avco arc-simulation programs, additional arc heads are in various stages of development and will be incorporated into the ROVERS facility as they become available. It is planned in the near future to have an arc head available on the ROVERS facility to cover the pressure range of 10 torr to atmospheres.

2.3 STRUCTURES

2.3.1 Reference Design Performance and Technology Development Requirements

The structural development test plan represents the minimum types of tests required to obtain design information and verify performance to ensure that an efficient structural design is evolved.

The scope of the tests depends to a great extent on the criticality of the structural weight fraction. If there is an ample allowance in the capsule system for structural weight, and if conservative design practices may be used with large margins of safety in areas of uncertainty, the number of the tests can be minimized. If, however, weight restrictions require that more or less unconventional or untried methods be used for analysis, with small margins of safety, more extensive testing will be required to verify theoretical analyses and performance predictions.

The development plan is divided into two categories: tests for design information and tests for performance predictions. The division depends to a degree on whether a component can be treated separately or has a major interaction with a non-structural element.

The major design requirements for development tests occur in the entry shell structure. The reference design consists of a honeycomb sandwich conical shell stiffened by a ring at the forward and aft end with another integral ring serving as a hard point for attachment of the payload.

There are many possible modes of failure for the sandwich shell and honeycomb core in which specific test data is lacking - for example; the general instability of conical shells is based on test data obtained for homogeneous isotropic cylinders. The edge restraints in the tests also do not simulate the actual elastic restraints that occur in the reference design. In addition, core strength requirements for the design were determined using data obtained from tests of flat plates and columns.

In other areas, such as the internal structure, numerous assumptions have to be made in order to reduce the size of the analysis effort. In this relatively complex structure, in some cases, it will be more economical in time and cost to test rather than analyze in detail a subcomponent.

The analysis of the structure for dynamic launch environments generally uses a combined analytical and experimental approach. A mathematical dynamic model of the structural system is developed and then modified by the results of vibration tests. The improved mathematical model is then used to predict the response of the structure to other dynamic environments.

The tests described therefore range from obtaining data which is not presently available for a certain class of structures to more or less conventional tests which directly support the design effort. A summary of the structural tests recommended is shown in Table XII. A more detailed description of the recommended tests is given in Table XIII and described in the sections following.

2.3.2 Entry Shell Structure

2.3.2.1 Design Information Acquisition for Entry Phase

1. Conical Shell-End Ring Stability -- Published test data for buckling of conical and cylindrical shell is applicable only to shells whose edges are either simply supported or clamped. The configuration of the blunt-cone entry shell requires that the outer edge of the shell be suitably supported so that the full strength of the shell corresponding to at least a simple support condition can be developed. An analysis was developed which estimates the stability of the ring-shell combination assuming that inextensional deformation is applicable. Since the resulting ring weight can constitute a significant fraction of the shell weight, verification of the analysis is required before a configuration freeze.

The objective of the test will be to verify the theoretical analysis of the shell-ring stability. These tests will be conducted over a range of cone angles, relative stiffness of ring and shell, and pressure distributions both symmetrical and unsymmetrical. Before conducting the shell-ring tests, measurements are necessary to determine the stiffness of the rings since there will be a reduction of the effective moment of inertia of the cross section due to the curvature of the rings. This effect can be predicted for rings of circular or rectangular cross sections, but no analysis is available for arbitrary cross-section geometry.

The boundary conditions of the shell-ring structure are very important considerations in the stability tests. The external pressure must be developed on these shells without imposing any external restraint or force on the elastically supported edge. This condition could be achieved by supporting the conical shell at the small forward end in a subsonic wind tunnel. The large end would then be free to deform into the buckling pattern associated with minimum pressure. The wind tunnel would be required to develop a differential of approximately 2 lb/in^2 between the external and internal pressure.

To verify the theoretical analysis, a quarter-scale monocoque shell structure would be adequate.

TABLE XII

STRUCTURAL TEST SUMMARY

Item No.	Primary Operating Function of the Element(s)	Critical Area	Description of Elements Subassembly, Etc.	Phase of Mission of Concern	Design Information	Performance Prediction	Flight Test	Mfg.
							Design Verification Required	Concepts & Methods Test
1	Structural Support	Yes (stability)	Conical shell end ring	Entry	X			
2	Structural Support	Yes (stability)	Honeycomb sandwich	Entry	X			X
3	Structural Support	Yes (dynamic response)	Entry Shell	Entry	X			
4	Thermo-Structural	Yes (mechanical integrity)	Ablator/Bond/Structure	Sterilization and Spaceflight		X		
5	Thermo-Structural	(Mechanical Integrity)	Ablator/Bond/Structure	Entry		X		
6	Structural Support	(selection of efficient structural details)	Fittings, Joints, Attachments & Bonds	Assembly to Mars impact	X			
7	Thermal Protection	Yes (mechanical integrity)	Ablator	Sterilization, Spaceflight, Entry	X			
8	Structural Support	Stiffness and damping characteristics	Suspended structure	Ground handling, launch, Entry, parachute deployment	X			
9	Structural Support	Overall performance prediction	Suspended structure	Ground handling, launch, entry, parachute deployment		X		

TABLE XIII

STRUCTURAL DEVELOPMENT TESTS

Element	Mission Phase of Concern	Design Analysis Problem Area or Requirements	Test Objectives	Test Description	Test Conditions Desired	Typical Test Facilities & Equipment
Entry Shell Structure	Entry	Conical Shell-End Ring Stability	Verify the theoretical analysis of the shell-end ring stability	Measurement of stability as a function of relative stiffness of shell and ring, pressure distribution.	1. Force free boundary conditions at large diameter. 2. 1/4-scale or less 3. Stiffness simulation of full-scale structure not required 4. Differential pressure less than 5 lb/in.^{2}	1. Subsonic wind tunnel
	Entry	Honeycomb Sandwich Conical Entry Shell stability	Verify theoretical analysis and obtain empirical design data for failure modes including general instability, intracell buckling, face-sheet wrinkling, core crushing, and core shear.	Measurement of the stability of honeycomb sandwich shell as a function of face-sheet thickness, core depth, core material, and cell size.	1. Hydrostatic pressure plus axial tension 2. 1/4-scale or less 3. Stiffness simulation not feasible due to gage limitation 4. Differential pressure less than 5 psi.	1. Pressure differential developed by partial vacuum in interior of shell 2. External forces applied by hydraulic jacks through load cells
		Entry Shell Structure Dynamics	Verify method of predicting frequency and mode shapes of entry shell structure	Measurement mode shapes and frequencies of conical shell structure as a function of shell stiffness, ring stiffness, and honeycomb core depth	1. Simulated boundary conditions associated with actual flight conditions 2. Frequency from 0.5 to 1000 cps	1. Electromagnetic shaker 2. Air jet shaker
	Sterilization and Spaceflight	Compatibility of ablation and substructure	Verify predicted strains, stresses deflections and margins of safety	Full-scale ablator and structure subjected to sterilization process and cold soak environment	1. Simulated restraints and supports of adjacent structure	1. Oven 2. Space chamber
	Entry	Thermal stresses due to entry temperature gradients	Verify predicted strains, stresses, deflections and margins of safety	Full-scale entry shell with temperature gradients simulated by quartz lamp radiant heaters	1. Critical temperature gradient combined with aerodynamic loading	
	Manufacturing	Selection of manufacturing process to minimize imperfections which will reduce strength of entry shell				
Joints, Fittings, Attachments and Bonds	Complete Mission	Selection of efficient structural details	1. Determine margin of safety for design concepts under combined and sequential loading 2. Provide design data such as stiffness characteristics for static and dynamic analysis	Specific tests will be designed to simulate critical loading for each particular component	Static, oscillatory and shock-loading as determined by prior analysis.	
Ablator	Sterilization	Effect of sterilization cycle on mechanical properties of ablator	Provide design data for analysis of ablator and structural compatibility	Measurement of mechanical properties after exposure to sterilization cycle: a) Stress-strain curve b) Thermal strain c) Dimensional stability	1. Simulated sterilization cycle	
	Spaceflight and Entry	Effect of spaceflight vacuum and thermal cycling on mechanical properties	Provide design data for analysis of ablator and structure compatibility and integrity during spaceflight and entry	Measurement of mechanical properties after exposure to spaceflight vacuum and thermal cycle.	1. Prolonged exposure to simulated spaceflight vacuum environment 2. Thermal cycle as predicted by analysis 3. Measurement made both in vacuum and ambient pressure	
Suspended Structure	1. Ground handling 2. Launch 3. Entry 4. Parachute Deployment	Stiffness and vibration characteristics required for use in static and dynamic analysis	Provide design information	1. Measurement of static influence coefficients 2. Measurement of natural frequencies of components	1. Static-force deflection measurement at critical locations determined by analysis 2. Sinusoidal frequency sweeps	
	1. Ground handling 2. Launch 3. Parachute Deployment 4. Entry	Performance of Complete suspended structure	Verify performance prediction of suspended structure	1. Simulated external forces, moments and inertial reactions applied to structure at critical locations determined by prior analysis 2. Simulated vibration input at critical locations	1. Critical static forces to be determined 2. Parachute deployment 18,000 pounds 3. Vibration 20 g's 2 to 50 cps 1.5 g's 50 to 300 cps	

2. <u>Honeycomb Sandwich Conical Entry Shell Stability</u> -- The present analysis of the sandwich wall entry shell utilizes theory developed for homogeneous isotropic conical shells. The analysis assumes that the core is rigid and uses the concept of effective wall thickness and effective Young's modulus when applying the homogeneous isotropic theory experimental results. The core density, which is proportional to its strength, was chosen to be no less than 3 percent of the density of the face-sheet material. This conservative value, based on test results of plates and columns, was selected to account for the various types of failure modes associated with honeycomb sandwich construction. These failure modes can be defined as intracell buckling, face-sheet wrinkling, core buckling and crushing, and core shear. In addition, there is the possibility of coupling of one of these modes with a general instability failure of the shell.

Since the honeycomb core represents a significant fraction of the total shell weight, a decrease in the density ratio would result in a meaningful weight reduction. To accomplish this, applicable test data is required relative to the failure modes described above.

Another related problem, for which sufficient test data does not exist, is the allowable minimum face-sheet thickness for a sandwich structure which will give repeatable test data. A weight reduction of the shell structure of up to 40 percent could be achieved if the face-sheet thickness could be reduced to the theoretical value required for strength and stability.

The present analysis shows that the axial tension in the shell aft of the payload has a small influence on the stability of the shell. There is presently no data to confirm this analytical conclusion. The objective of the tests is to verify theoretical analyses and to obtain empirical design data where theoretical analyses are not available or feasible.

Honeycomb sandwich construction parameters such as cell size, core material, face-sheet thickness and core depth, will be varied. The shell parameters will include cone angle, diameter and slant length.

The majority of the structural shell tests would be conducted using a hydrostatic pressure. Data could be obtained on 1/4-scale models using a differential of less than 5 lb/in^2 between the external and internal pressure. Precise geometrical scaling will not be feasible due to minimum core depth and sheet thickness limitations.

To verify that axial tension has a negligible interaction with the hydrostatic critical pressure, tests should be conducted on identical shells with identical external pressure with the axial force reacted at either the small or large diameters.

3. Entry-Shell Structure Dynamics -- The critical dynamic loading on the entry-shell structure is expected during entry of the flight capsule. This dynamic loading is associated with the rigid body motions of the capsule whose frequencies are generally less than 6 cps. Calculations of the natural frequencies have shown that typical entry shell structures have frequencies as low as 8 cps occurring in the second or third harmonic. Coupling of the structural dynamics does not appear to be a problem since the major component of the unsymmetrical external pressure loading occurs in the first harmonic; this corresponds to a higher structural frequency than the second and third harmonic. However, the analytical data used in the analyses have not been fully verified by experiment. Boundary conditions, payload-shell attachment structure, and the base-ring structure also affect the accuracy of the analytical solutions. The analyses also use the concept of equivalent thickness and Young's modulus in order to apply the techniques developed for homogeneous isotropic structures to honeycomb structures. The validity of the methods and results have not been demonstrated experimentally.

The frequencies calculated also assume that the shell is unstressed. Analyses of idealized shell structures have shown that the natural frequency of shells are reduced when stresses approach the critical buckling stresses. The ratio of applied loading to the critical buckling load at which this effect becomes important should be verified by experiment.

The objective of these tests is to verify the basic analytical methods of predicting the response of the conical entry shell structure to dynamic entry loads by measuring the frequencies and mode shapes of monocoque and ring-stiffened shells.

To determine the effect of shell stresses on frequency, tests should also be made of shells under surface pressure loading. Tests should then be made on honeycomb sandwich shells to determine the validity of analytical techniques when extended to sandwich shells.

For the majority of the tests the shells would have simulated boundary conditions associated with actual flight conditions. The shell modes will be excited by either electromagnetic shakers or air jet shakers.

To determine the affect of prestress on the natural frequencies, an external surface pressure will have to be applied which will probably preclude the simulation of flight-boundary conditions. The applied pressure will be less than 5 lb/in^2.

2.3.2.2 Performance Prediction for Sterilization and Space-Flight Phases

The necessity of full-scale performance tests for the compatibility of the ablator and substructure during the sterilization heat cycle and the subsequent space-flight temperature distribution depends primarily on the selected ablator material and the expected temperature ranges. If the results of the mechanical property evaluation tests of the ablator indicate that the margin of safety throughout the mission sequence are large, these tests would not be required. However, if performance predictions tests of the thermal control system are required, then measurements of strains and deflections during these tests would be worthwhile.

The objective of these tests is to verify predicted strains, deflections and margins of safety during the sterilization heat cycle and the subsequent space-flight temperature distribution.

The full-scale heat shield will be subjected to the sterilization heat cycle and subsequent space-flight temperature environments. The structure will be supported in a manner simulating the restraints of the internal structure and afterbody.

2.3.2.3 Performance Prediction for Entry Phase

The necessity of full-scale performance tests of the heat shield with simulated entry temperature gradients depends on the mechanical properties of the ablative material and margins of safety predicted in the thermal stress analysis of the composite heat shield structure.

The objective of the tests is to verify performance predictions of the thermal stresses and displacements in the ablator and substructure due to entry temperature gradients.

Temperature gradients due to entry heating will be developed by quartz lamp radiant heaters. To simulate the actual gradients as a function of time, the thickness of the ablator will be reduced if necessary, and the power to the lamps will be controlled by a programmed feedback controller.

2.3.3 Joints, Fittings, Attachments and Bonds

2.3.3.1 Design Information Acquisition for Complete Mission

The flight capsule structure has many joints, fittings, attachments and adhesive bonds which undergo many combinations of loading in combination and in sequence, both static and dynamic. Many of these components can be analyzed with various simplifying assumptions; however, in many cases, the more efficient procedure would be to test the design concepts after a preliminary analysis has been performed rather than conduct a detailed analysis. This is particularly true if the design criteria for a particular component is deflection or permanent deformation. The need for tests of a specific component will depend on how critical the weight of component is, i.e., if a large margin of safety is acceptable. In other cases, the physical size or cost of a component could be critical.

The objective of these tests will be to determine the margins of safety for design concepts under combined and sequential loadings. Degradation of a joint or attachment due to preceding environmental conditions will be determined. Stiffness characteristics for use in both static and dynamic analyses will be determined when necessary.

A detailed list of joints, fittings and attachments and loading conditions cannot be formulated at this point in the preliminary design.

Static, oscillatory and shock loading will be applied to the components while simulating the bending, direct and shear stresses as determined from prior analysis. The temperature of the components will correspond to the predicted operating temperatures when it has a critical influence on performance of the component.

2.3.4 Heat Shield Ablator

2.3.4.1 Design Information Acquisition for Sterilization Phase

The function of the ablator material is primarily to provide thermal protection for the structure and internal components of the flight capsule. The ablator contributes negligible strength to the primary capsule structure and is generally neglected in analysis of the static structural response to external loading. The principal structural requirement for the ablative material is merely to be compatible with its supporting structure throughout the mission profile unit entry shell ejection.

The ability of the ablator to be compatible with a specific substructure is reflected in its relative thermal strain and stiffness properties as compared to the substructure material and its ductility or strain to failure. The compatibility has to be considered over the expected operating temerature range, but primarily at the lowest expected temperature.

The above mentioned mechanical properties which are required in the design calculations, can be altered by the sterilization heat cycle, a result of possible additional curing of the material. This post curing could also affect the dimensional stability of the ablative material causing either permanent expansion or shrinkage of the material. These changes in dimension could induce residual compressive or tensile stresses in the composite heat shield structure.

The mechanical properties of the ablative material will be measured after exposure to the simulated sterilization heat cycle to provide data for the design analysis. The principal structural properties which will be measured will be the stress strain relationships, thermal strain, and ultimate tensile stress and strain as a function of temperature.

The samples will be exposed to the sterilization heat cycle and measurement made of mechanical properties.

2.3.4.2 Design Information Acquisition for Spaceflight and Entry Phases

During spaceflight, if the ablator is exposed to the deep-space vacuum, the mechanical properties can be altered by outgassing of some of the volatile constituents. Combined with the exposure to vacuum is the possibility of thermal cycling inducing a low-cycle fatique-type of failure. The thermal cycling is induced by periodic exposure to solar radiation.

The mechanical properties of the ablative material will be measured after prolonged exposure to the simulated space environment to provide data for design analysis.

The duration of the prolonged vacuum exposure will depend on the expected mission profile. Methods of accelerated vacuum exposure such as the use of elevated temperatures will be utilized when feasible. Tests will be performed on ablator specimens alone and composite ablator structure specimens. The test will be conducted at 10^{-5} torr or less.

Measurements of mechanical properties should also be made.

2.3.5 <u>Internal Structure</u>

2.3.5.1 Design Information Acquisition for Ground Handling, Launch, Entry and Parachute Deployment

Stiffness characteristics are required for static and dynamic analysis of the internal structure when subjected to the ground handling, launch, entry and parachute opening loads. These quantities can be predicted analytically for preliminary analysis subject to simplifying assumptions. For final design, it is necessary to know the value of these quantities with greater accuracy as well as measurements of damping of the structural system, a quantity which is very difficult to predict analytically.

Measurements of static influence coefficients of the internal structure will be made to provide data for the static and dynamic design analysis. Natural frequencies of system components will also be measured to be used in developing equivalent dynamic models for use in the dynamic analysis.

Static force deflection measurements will be made at critical locations determined by analysis, to be used as structural influence coefficients. Sinusoidal frequency sweeps will be made of principal components to determine natural frequencies for use in developing equivalent dynamic models.

The number of measurements required for each mission phase will depend on the significance of the change in the structural configuration in each subsequent mission phase.

2.3.5.2 Performance Prediction for Ground Handling, Launch, Parachute Deployment

The internal structure is a complex redundant structure composed of frames, trusses, shear webs and shells. The response of the system to static loads can be conservatively predicted using computerized analytical methods and experimental data obtained from tests described in paragraph 2.3.5.1. The need for verification of the results of the analysis depends on the degree of conservatism permissible in the completed design. The permissible conservatism is reflected primarily in the allowable weight. If weight is a critical problem in one of the mission phases, it will be necessary to simulate the loading during that phase on the structural assembly to verify the design calculations. At this point in the design cycle, it is not possible to state with certainty the need for a specific test to verify a static performance prediction.

The state of the art in dynamic analysis is not as far advanced as static analysis, hence a confirmation of performance prediction is necessary for the ground handling, launch, entry and parachute deployment mission phases.

The static performance prediction will be verified by applying equivalent forces, moments, reaction and restraints as determined by analysis. These loadings will be applied in sequence if necessary.

The dynamic performance prediction will be verified for the ground handling, launch and entry mission phases by comparing the results of sinusoidal and random vibration inputs to analytical predictions using the design dynamic model.

The parachute dynamic loads performance prediction will be verified by equivalent shock loading obtained from drop test or other suitable methods.

2.4 THERMAL CONTROL

2.4.1 Reference Design Performance and Technology Development Requirements

The thermal control system function is to maintain the temperature levels of the various components of the flight capsule within prescribed limits.

A secondary objective is to provide pre-entry temperature levels for the entry shell to minimize their weight while maintaining their thermo-structural compatibility. Finally, the temperature control system should minimize flight spacecraft temperature excursions after separation as well as minimizing spacecraft power requirements before separation.

The thermal control is basically a passive system augmented by heating elements placed at required locations. It is incorporated by applying coatings and utilizing the existing structural members for heat flow management and heat leakage control; local insulation may also be required. The thermal control development problems and technical requirements for either the entry shell or canister subsystems are similar (low emissivity coatings are required for both); however, the canister subsystem is simpler since the coating would be applied to a metallic substrate. For the entry vehicle shell the substrate is organic and presents outgassing, potential low temperature (below transition phases), and decontamination/sterilization problems. Thus, even though discussed for the entry vehicle shell only, the discussion is also applicable to the sterilization canister. The thermal control system consists of the coatings, insulation materials, and heaters. The radiative heat interchange between the internal surfaces is controlled by proper surface conditioning, and convective heat transfer is usually negligible although it may depend on the prevailing g - level and the degree of internal pressurization.

Since the flight capsule is primarily in the shade of the spacecraft, a low emissivity ($\epsilon \approx 0.05 - 0.1$) coating is essential. The operating temperature limits of the components appear to be quite compatible with the lower temperature heat shield limit (approximately -100° F).

Of importance are their optical and adhesion characteristics when applied to the substrate (the composite of coating and heat shield) and exposed to the environments. This in turn will require effort in the application methods development (bonding and surface preparation) to assure adhesion and proper coating thickness selection, and determination of the properties of the composite that are required to assure the performance. The effect of ETO decontamination and dry-heat sterilization cycles and the effect of low temperature and vacuum during cruise and orbit near Mars on the coating composite performance will have to be determined.

TABLE XIV

TEST SYNTHESIS SUMMARY - THERMAL CONTROL

Item No.	Primary Operating Function of the Element(s)	Critical Area	Description of Elements, Subassembly, etc.	Phase of Mission of Concern	Design Information	Performance Prediction	Concepts & Methods Test
1	Passive temperature control	Yes (performance data)	Thermal Control Coating/Heat Shield Composite	Cruise, planetary orbit and post separation	X	X	X
2	Thermo-structural	Yes (for survival of environment and use of properly conditioned material properties)	Thermal Control Coating/Heat Shield Composite	Decontamination and Sterilization	X	X	
3	Component Insulation	No (but "in place" property information needed)	Thermal Insulation Materials	Cruise, planetary orbit, post separation, entry, parachute descent	X		
4	Insulation	Same as Item 2	Thermal Insulation Materials	Decontamination and Sterilization	X		
5	Active Temperature Control	Yes (integration & performance)	Electric Heaters	Cruise and planetary orbit		X	X
6*	Active Temperature Control	Same as Item 2	Electric Heaters	Decontamination and Sterilization			
7	Local heat flow control and structural	Yes (performance data)	Separation mechanisms, heater interface, structural joints, payload components/support structure interfaces, etc.	Cruise, planetary orbit, post separation, entry, parachute descent	X	X	X
8*	Thermo-structural	Same as Item 2	Separation mechanisms, heater interface, structural joints, payload components/support structure interfaces, etc.	Decontamination and Sterilization	-	-	-
9	Overall heat flow management and thermal structural	Yes (overall performance verification)	Entry vehicle including suspended capsule	Planetary orbit, post separation entry, parachute descent	-	X	
10*	Thermo-structural	Same as Item 2	Entry vehicle including suspended capsule	Decontamination & Sterilization	-	-	

*It is assumed that in the sterilization program the effect of decontamination and sterilization on components will be determined.

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Prediction of the heat-flow and temperature distributions in the entry vehicle shell and payload modules is quite feasible; however testing is required because of the uncertainties of actual contact resistances present in the many joints and interfaces and because of the many interacting radiative paths.

Consideration of thermo-structural compatibility per se (during spaceflight) may require model testing (depending on safety margins allowable). A half-scale model could be used. However, full-scale structural models may be available for other purposes.

The critical areas of the development are associated with the behavior of the composite of coating and heat shield, and the determination of heat-flow patterns through the structural members and joints to the payload for all environments anticipated. The degree of the severity of the problems depends on the allowable emissivity and temperature variation and structural safety margin, while skillful control may contribute to a decrease in thermal protection weight.

Facilities for testing present no critical problems even for full-scale tests. The tradeoff between full-scale and half-scale testing will reduce itself to a time and cost consideration, provided the half-scale model is sufficient for design purposes.

A summary of the recommended development tests is given in Table XIV.

2.4.2 Thermal Control Coatings and Composites

The analysis of the thermal control requirement indicated the importance of the thermal interface with the spacecraft in terms of the relative geometrical configuration and coatings to be used on the sterilization canister and the entry vehicle.

The problem appears in the determination of the optical performance limits of the coating and its ability to withstand the various environmental extrema while adhering to the substratum. Thus the objective of the development tests is a) establishment of the method of application to the substratum, and b) determination of the emissivity variations or changes resulting from the effect of the substratum, and limits of degradation (if any) due to that effect, or the decontamination, dry heat sterilization, vacuum, cold-soak and other environments (in composite form).

Since a variety of coatings with the required optical characteristics are available, and reliable surface treatment and coating methods are known today for the application to metallic surfaces, the major development problem will exist in the determination of the performance of the thermal control coating/heat shield composite.

TABLE XV

THERMAL CONTROL COATING/HEAT SHEILD COMPOSITE SCREENING TEST PRECONDITIONED (DECONTAMINATED AND STERILIZED)*

Element	Mission Phase of Concern	Design Analysis Problem Areas or Requirements	Test Objectives	Test Description	Test Condition Desired	Typical Test Facilities Equipment and Procedure
Thermal Control Coating/ Heat Shield Composite	Cruise, planetary orbit and post separation	Selection of thermal control coating which maintains its performance characteristics during exposure to space flight conditions when applied to substratum	Provide basic characterization leading to selection of coatings and a method of coating application	Testing after exposure of coating substratum samples to simulated spaceflight environments	1. No. of application methods not to exceed 3 2. No. of coatings not to exceed 10 for each method 3. Environmental test parameters or their derivatives to approach the design operating conditions 4. No. of tests will depend on reliability requirements 5. Vacuum: 10^{-6} Torr (min) 6. Solar simulation: Mars intensity	Thermal cycling test facility and solar flux cycling facility
			a) Determine mechanical compatibility with the substratum	Measurement of various mechanical and adhesion characteristics	As required for preliminary evaluation	
			b) Determine Optical Stability	Measurement of solar absorptance Measurement of spectral emittance Measurement of total solar absorptance/ emittance	As required for preliminary evaluation	Gier-Dunkle Solar reflectometer Beckman recording Spectometer
	Decontamination Sterilization	Changes in material decomposition and behavior and adhesion charactersitics	Select materials which are compatible with decontamination/sterilization cycles	Measurement of mechanical, adhesion and optical properties of preconditioned samples (See above)	See above (1 to 4). Decontamination and Sterilization cycle as specified	Laboratory furnace (Standard equipment)

*Unless otherwise noted

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2.4.2.1 Design Information Acquisition

The thermal control coating composite's control function begins with the earth orbit phase and ends at onset of reentry. Its performance, however, is affected by all prior phases. Those of major concern are reflected in the test conditions required for determination of design information as shown in Tables XV and XVI. The specific test objectives, brief description of the tests, test facilities and equipment required are included. First the optical properties, α and ϵ, should be determined since the flight capsule is permanently oriented to space during cruise and only limited power from the spacecraft is available, low ϵ-values of the sterilization canister will be required for effective thermal control of the entry vehicle. Low ϵ-values are characteristic of surface preparations of metals or coatings of a metallic nature. These properties will be determined for the composite previously exposed to space environments, and for further exposures until the coating function ceases. The required characteristics of the coating itself is that the porosity be sufficient to allow outgassing to occur without damage to the coating, in the composite it must maintain stability, when exposed to large temperature fluctuations and a decontamination environment; and it must maintain thermo-mechanical compatibility with the ablator under environmental temperature extremes. Thus porosity and permeability of the coating will be determined as well as the pertinent mechanical and surface properties (bond strength, flexibility, fatigue and abrasion resistance) as re-required for the various prior exposures and operating conditions during spaceflight.

Two series of tests are contemplated, the initial or screening tests are of somewhat limited scope in acquiring properties and will be run concurrently with the initial evaluation of coating techniques and methods (See Table XV). In the second series of tests the more promising composites and coating methods will be evaluated (See Table XVI).

2.4.2.2 Test Facilities, Equipment and Operation

The test facilities, equipment and techniques to be utilized in achieving objectives of this phase of the developement program are described below by test categories.

1. Measurement of Solar Absorptance (α) -- The optical measurements use the Barnes Model R4-F2 radiometer and Bettman extended-range ratio recording spectro-photometer. In addition, a Gier-Dunkle Solar Reflectometer is used for the measurement of solar absorptance.

2. Spectral Emittance, ϵ -- The spectral emittance of materials is measured by a method similar to that used by Gier, Dunkle, and

TABLE XVI

THERMAL CONTROL COATING/HEAT SHIELD COMPOSITE DEVELOPMENT TEST
Preconditioned (Decontaminated and Sterilized)*

Element	Mission Phase of Concern	Design Analysis Problem Areas or Requirements	Test Objectives	Test Description	Test Conditions Desired	Typical Test Facility Equipment & Procedure
Thermal Control Coating/ Heatshield Composite	Cruise, planetary orbit and post-separation	1. Determination of the temperature response of the system to the spaceflight environment and maintainance of components within prescribed temperature limits 2. Preparation of thermal control coating specifications	1. Provide characteristic parameters for design use and performance prediction	Testing after exposure of coating/substratum samples to simulated spaceflight environments	1. Number of application methods not to exceed one per coating. 2. Number of coatings not to exceed 2. 3. Environmental test parameters or their derivatives to approach the design operating conditions 4. Number of tests will depend on reliability requirements	
			a) Determine optical properties of the coating on substrate	Measurement of total absorptance/ emittance to solar sources Measurement of solar absorptance Measurement of spectral emittance	Exposure to solar simulation at Mars intensity, thermal and solar flux cycling prior to measurement	
			b) Determine degradation and stability after exposure to various environments (determine various physical properties)	Measurement of density Measurement of porosity Measurement of Permeability Photomicrography and Xrays Ultrasonic testing for voids & inclusions Measurement of abrasion resistance	Thermal cycling in High Vacuum and Solar Flux Cycling in vacuum of unexposed and exposed samples. Vacuum: 10^{-6} Torr. min. Mars intensity solar simulation as required	
		3. Mechanical compatibility with the substratum	Provide information to assure mechanical compatibility with the substratum	Measurement of various mechanical and adhesion characteristics a) Bond Strength (Shear and Tensile) b) Flexibility c) Fatigue Resistance	As required for evaluation. Same as for other physical properties above.	
	Miscellaneous Environments	Assurance of performance and satisfaction of specifications	As required by Government Specifications --------------------------			
	Decontamination/ Sterilization	Assurance of performance after exposure to ETO and dry heat.	Provide property parameters for design use and thermal control performance prediction ("as exposed" condition)	Sample exposure to ETO and dry heat (See tests for spaceflight phase	Sterilization Cycle as specified and screening program -------------)	Laboratory furnace (Standard Equipment)
	Manufacturing	Assurance of reproducibility, homogenity and integrity of materials during exposure to various environments	1. Examine raw materials for chemical and physical variations determine effect on outgassing 2. Select and evaluate component for maximum bond strength 3. Specify and scale-up application process 4. Develop nondestructive test techniques 5. Verify application Method	Depends on the screening test results	and selection of reference material.	

*Unless otherwise noted

Bevans, (Reference 1) with a Gier-Dunkle infrared reflectometer together with a Perkin-Elmer Model 98 monochromator and associated control and recording equipment.

3. Thermal Cycling Tests in Ultra-High Vacuum -- The purpose of these tests is to evaluate the coating adhesion/flexibility of the substrate-coating composite during thermal cycling in an ultra-high vacuum. Two types of tests are to be performed. One test consists of cycling the composite sample between the hot-soak and cold-soak temperature limits. The other test consists of exposing the composite to an incident solar flux level cyclically. Both tests are described below:

a. Temperature Cycling Tests -- The purpose of the test program is to evaluate the effects of hot and cold cycling at high altitudes on thermal control surface coatings. The evaluation should determine the stability and adequacy of the surface coating to withstand temperature cycling within specific limits and at high vacuum 1×10^{-6} torr).

Cold-and hot-soak panel tests have been employed extensively in the past to evaluate design suitability at the lower and upper limits of the thermal protection system temperature envelope. The thermal control system performance and the effect of bond, basic ablator, and joints on the coating performance represent the major areas of investigation.

The specimens (10 x 10 sample size) are mounted in holders so that the face (coated surface) is a quarter of an inch from the heat transfer surface. The chamber will be pumped down to a pressure of 1×10^{-6} torr using liquid nitrogen in the cryopanels.

The test samples will then be subjected to the various optical and physical tests to characterize the coated surface before and after each test.

b. Solar Flux Cycling Tests -- The tests of the coating effectiveness should be performed in a space-simulation chamber equipped with a solar simulator such as the Avco 200 liter Space Simulation Chamber.

A 5-kw high-pressure Xenon lamp is used as a solar radiation simulator. It has been found that in practice it is necessary to supplement the ultraviolet radiation of the high-pressure Xenon lamp to obtain a satisfactory intensity in this region of the zero-air mass solar spectrum. High-pressure mercury lamps (type GE AH-6) are used for this purpose and one is mounted in a parabolic reflector just to the side of the main Xenon lamp.

[1] Gier, J. T., R. V. Dunkle, and J. T., Bevans, Measurement of Absolute Spectral Reflectivity from 1.0 to 15 Microns, J. Opt. Soc., 44 No. 7 pp 558-562 (July 1954).

4. <u>Measurement of Mechanical and other Physical Properties</u> -- The mechanical and other physical properties of the coatings will be studied as discussed below:

a. Bond Strength -- Both shear and tensile bond strengths of the coating to the substructure will be evaluated before and after exposure to simulated environments. Test specimens will consist of the coated substrate material bonded at the coating surface with a high strength adhesive to a steel plate. Proper specimen design and testing in a universal test machine will generate either shear or tensile loads at the coating-substrate bond line. Tests will be conducted at the various tests speeds and temperatures of greatest interest.

b. Flexibility -- The coated substructure material will be tested on a universal test machine to determine the elongation behavior of the coating under conditions of varying test speeds and temperatures. These tests will be carried out before and after environmental exposure.

c. Fatigue Resistance -- The fatigue resistance of the supported coating will be tested in bending at low and moderate frequencies to simulate ground and launch vibration. Test conditions will be determined by the specific environments. Specimens will be instrumented so that either cracks in the coating or loss of bond will signal the end of the test, whichever is determined to be critical.

d. Abrasion Resistance -- Since surface roughness can effect the optical properties of many coatings, the abrasion resistance of the coating will be evaluated. The test will be similar to the Tabor test, where a wheel of specific roughness abrades the coating at a specific speed, and resulting surface roughness and weight loss are measured. Optical properties will be measured before and after abrasion.

e. Permeability -- The permeability of the supported or unsupported coatings will be measured by simple gas-flow tests. The pressure drop across the specimen will be measured at specific applied pressures. The sprayed coatings can be tested directly on the substrate material if the permeability of the substrate is known. Sheet coatings can be tested either free or as applied to the substrate.

f. Density and Porosity -- The bulk density of the coating in either the supported or unsupported state will be measured by bulk measurements of size and weight, or with the mercury porosimeter. The coatings applied as sheets can be tested unsupported. Where bulk measurements are impossible, the mercury porosimeter will be used. The apparent density will be measured with a Beckman Air Comparison Pycnometer. The difference between bulk and apparant densities is a measure of the open porosity of the sample. With nonporous materials, air pycnometer measurements will give bulk density data.

g. Miscellaneous Testing -- The coated surface will be photomicrographed in life size, photomicrographed at the centroid of the areas, and measured for both surface flatness and surface roughness, X-rays will be made normal to the polished surface and parallel to the lamination, and finally ultrasonic tests will be conducted to determine the presence of voids and inclusions.

Other standard ground environmental tests will have to be conducted to evaluate the effects of climatic environments on ablator surface coatings. The evaluation of the ablator surface coatings shall, in general, determine the stability and adequacy of the working surface coating to withstand exposure to salt spray, sand and dust, and temperature-humidity environments per Federal Specifications.

2.4.3 Thermal Insulation Materials

Insulation may be required in areas where thermal control by limiting or increasing of the heat leakage from external surfaces to the interior of the flight capsule or between adjacent components is required. The pre-selection and specification of insulation materials, in particular, low-density foam and complex-multi-layer superinsulation blankets, will require thermal property measurements under actual environmental conditions as well as tests to verify their mechanical behavior. Wherever possible, "off-the-shelf" products will be used, thus not entailing a specific development effort. The problem is not considered critical in the sense that relatively simple shapes, reasonably small temperature differences at moderately low temperatures have to be maintained. However, design information (properties) will have to be verified by tests in the environment, since the insulation characteristics depend largely on the method of application and may be affected by the sterilization - decontamination process. Procedures and process controls must be developed to ensure reliable and reproducible performance. The thermal model test will supply the needed verification of performance prediction. The tests are summarized in Table XVII (for screening of insulations for decontamination - sterilization) and Table XVIII.

2.4.4 Heating Elements

The design studies indicate that heaters are required in various elements to provide additional thermal energy. The integration of heating elements into a component, in general, should not create a critical problem. In the case of the entry shell structure, however, the problem of assembly and installation of the heaters may prove to be difficult.

The design data may be easily acquired, while the overall performance prediction will have to be verified under various environmental conditions as described in Table XIX utilizing full-size sections of the entry shell with heaters imbedded. Their performance will also be verified in the thermal model test.

TABLE XVIII

COMPONENT INSULATION DEVELOPMENT TEST

Insulation Material Preconditioned (Precontaminated and Sterilized)

Element	Mission Phase of Concern	Design Analysis Problem Area or Requirement	Test Objectives	Test Description	Test Conditions Desired	Typical Test Facility Equipment and Procedure
Component Insulation	Cruise to Impact	Provide "in place" performance information for insulation materials applied to (actual or dummy) components of concern	Provide characteristic parameters for design use and performance prediction	Measurement of thermal gradients within the insulation and across the component during exposure to a controlled temperature environment and vacuum and comparison with predictions	1) Number of materials: Not to exceed 2 2) Number of application methods not to exceed 1 for each material 3) Environmental test parameters or their derivatives to approach the design operating conditions 4) Number of tests will depend on the number of components considered and reliability requirements 5) Vacuum: 10^{-6} Torr (min) 6) Temperature range: -100 to + 300°F 7) Samples to be instrumented as required.	Environmental test chamber (Standard equipment)

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TABLE XVII

INSULATION MATERIAL SCREENING TEST

Preconditioned (Decontaminated and Sterilized)

Element	Mission Phase of Concern	Design Analysis Problem Area or Requirements	Test Objectives	Test Description	Test Conditions Desired	Typical Test Facility Equipment and Procedure
Insulation	Decontamination Sterilization	Selection and specification of efficient insulation materials after exposure to the decon-tamination and sterilization cycle.	Provide basic data (thermal and mech-anical) leading to materials selection, provide property parameters for design use and performance prediction.	Measurement of thermal conduc-tivity and mechanical properties before and after exposure to environment.	1. Materials considered: a) Super-insulation (multi-layer) b) Low-density foam 2. Number of brands not to exceed 10 for each material 3. Number of application methods not to exceed 2 for each material 4. Environmental test parameters or their derivatives to approach the design operating conditions 5. Number of tests will depend on reliability requirements	Guarded hot plate
	Miscellaneous Environments	Assurance of performance and satisfaction of specifications	As required by Government Specifications --------			

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TABLE XIX

HEATING ELEMENT INTEGRATION DEVELOPMENT TEST

Element	Mission Phase of Concern	Design Analysis Problem Area or Requirements	Test Objective	Test Description	Test Conditions Desired	Typical Test Facility Equipment and Procedure
Electric Heaters	Cruise and Planetary Orbit	Selection of Heaters and specification of heater integration	Provide characteristic parameter for specification, design use and performance prediction.	Measurement of thermal gradients across component or sample (x) with integrated heater during exposure to a controlled temperature environment and vacuum.	1) Vacuum: 10^{-6} Torr (Min) 2) Temperature Range: -100 to +300°F 3) Environmental test parameters or their derivatives to approach the design operating conditions 4) Number of tests will depend on the number of components considered and reliability requirements	Environmental test chamber (standard equipment)
	Contamination / Sterilization	1) Assurance of performance after exposure to ETO and dry heat	Provide property parameters (thermal and mechanical) for design use and performance prediction			
		2) Determine the performance of integrated heaters if used as internal heat sources during sterilization		Same as above but atmospheric environment		
	Manufacturing	Specify assembly techniques				
				(x) In particular full size section of the entry shell after exposure to ETO and sterilization cycle		

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2.4.5 Structural Joints and Thermal Contact Resistance

The structural members of the entry vehicle and canister provide the conductive path for the heat flow which accounts for a large portion of the integral heat transfer. The heat flow in continuous structural members may be easily predicted by analytical means; however, the transmission of heat through manufacturing joints between various elements of the vehicle, through bolted interfaces, clamps, and other interfaces which provide thermal contact resistances are a function of materials involved, contact pressure, surface finish, and environment. Contact pressure depends on torque applied to the bolts and forces acting on the joined surfaces. While torques may be specified, the acting forces must be calculated and will depend on the environment. In the reference design a "Mormon" clamp holds the entry shell attached to the suspended capsule; this type of contact will obviously affect the transfer of heat between these two elements. "Black boxes" are bolted to the payload modules, and the entry shell is attached to the canister base through an adpater structure, all of which may constrict the heat-flow patterns. Due to the large number of joints and unorthodox types of contact resistance it will be necessary to determine the values of the thermal resistance by experimental means under simulated operating conditions. Functionally, the dry-heat sterilization and spaceflight phases are of main concern as the contact resistance may affect the sterilization cycle (require additional heaters during that phase), and of course in spaceflight where they may obstruct the temperature regulation. The specification and necessary knowledge of contact resistance depends to a great degree on the time element involved. If transients are involved, as during maneuvers, and if large temperature gradients are likely to exist throughout the body, the problem becomes more critical; however if heating elements are distributed properly, the severity of the problem may be alleviated. On the other hand the selection of the location of the heater may depend on the knowledge of resistance. Also, if multiple conductive paths are provided, the problem becomes less critical. The tests are summarized in Table XX. To minimize the development cost and time, standard joints will be used where feasible. The final performance prediction verification will be made for the overall thermal test model recommended for the entry vehicle should a half-scale model be selected. The scaling laws will be verified (if necessary) by testing full-and half-scale models of critical joints. The extent of the tests will be largely dictated by the degree of the complexity of the entry vehicle assembly and manufacturing methods. It may be sufficient to obtain the information on contact resistance in the entry vehicle-thermal model tests.

2.4.6 Entry Vehicle Subsystem

The development tests for the various elements of the thermal control system were primarily aimed at acquisition of design information required for system evaluation and analytical prediction of performance required for

TABLE XX

THERMAL CONTACT DEVELOPMENT TEST

Preconditioned (Decontaminated and Sterilized)

Element	Mission Phase of Concern	Design Analysis Problem Area or Requirements	Test Objective	Test Description	Test Conditions Desired	Typical Test Facility Equipment and Procedure
Thermal Contact Resistances (Structural Joints) (x)	Cruise, Planetary orbit to entry	1. Predictability of heat-flow patterns through interfaces after assembly and exposure to environments.	Provide characteristic parameter for design use, performance prediction and specification (manufacturing)	Application of typical heating rates to one component of the joined sample and measurement of temperature gradients across the sample during exposure to a controlled environment (temperature and vacuum)	1) Full-scale or half-scale model tests (see Table 8) depending on component size and considerations in regard to the validity of scale model test results.	Environmental test chamber (standard equipment)
		2. Specification of materials and contact pressure for critical joints in order to control heat flow.			2) Environmental test parameters or their derivatives to approach the design operating conditions.	
	Decontamination/ Sterilization	Assurance of reproducibility after exposure to ETO and dry heat.	Provide characteristic parameter for design use and performance prediction.			
	Manufacturing	Specification of assembly techniques, materials surface conditions, etc., to assure performance and reproducibility.			3) Number of tests will depend on the number of components considered and reliability requirements. 4) Vacuum: 10^{-6} Torr (min) 5) Temperature range: -100 to +300°F	

(x)In particular separation mechanism entry shell and payload components/support structure mounts.

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initial design purposes. As noted previously, for simple configurations this is an acceptable procedure; however for complicated geometrics and joints subject to various modes of heat transfer and to temperature-dependent loads interacting among each other, the predictability of the parameter variation becomes questionable. While design fixes are possible and may involve only small weight penalties, and while use of heaters may alleviate the excessive temperature gradients which might exist in a purely passive system, it is very difficult to predict the remedy on a purely analytical basis without experimental verification. Thus to avoid local or generalized hot or cold spots in payload compartments or to avoid local stress areas, is is desirable to construct a thermal model which would verify the analytical predictions. Such a model should be constructed at a sufficiently early time in the schedule to permit checks of thermal balance calculations. Sufficient temperature and strain instrumentation should be provided.

The scaling requirements must be determined. The electronic components may be simulated by heaters. However, scaling down may involve difficulties of minimum gage structural members which may contribute to the distribution of actual loads; superinsulation scaling is always difficult. Time, costs, and availability of facilities are involved in the decision. The full-scale model offers the advantage of minimum effort in designing the model, and high degree of simulation; however it may be expensive to construct and call for a larger facility. It can be used for demonstration and specification of assembly procedures and may be available in almost finished form from the structural development test program. A half-scale model may not necessarily be less expensive to construct, but might by available sooner; facility operating costs will be lower and facilities more accessible (Reference 2) although facilities for full-scale tests are available.

No problem exists in the area of half-scale testing from a technical standpoint; the laws controlling thermal-scale modeling of a spacecraft are well understood and are being reduced to a state-of-the-are technique for steady-state conditions. For the transient case, the problems are more severe, but some basic work has been reported (Reference 3). On this basis it is concluded that a half-scale model would be acceptable, but the final decision should be deferred until the design reaches a more detailed stage and the economics are assessed.

Both types of tests are described in Table XXI as alternatives. Space chamber operation, equipment and techniques have been described in the environmental test literature (Reference 4). Some of the operating characteristics are shown in Table VIII.

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[2] Hnilicka, M. P., and K. A. Geiger, Feasibility of Simulating Interplanetary Space, The Journal of Environmental Sciences, pp 28 to 33 (June 1964).

[3] Vickers, V. M. F., Thermal Scale Modeling, Astronautics and Aeronautics, p 34 to 39 (May 1965).

[4] Northeastern University continuous Education Lecture No. 3 Space Environment - Space Chamber Operation, Donald E. Lee, G. E., Spacecraft Dept., King of Prussia, Pa.

TABLE XXI

ENTRY VEHICLE SUBSYSTEM PERFORMANCE PREDICTION

Element	Mission Phase of Concern	Design Analysis Problem Areas or Requirements	Test Objectives	Test Description	Test Conditions Desired	Typical Test Facility Equipment and Procedure
Entry Vehicle	Planetary orbit and post separation	Performance prediction confirmation of thermal balance calculations and study of the effects of design configuration Note: See also comments on structures test for deformation design requirements.	Provide characteristic parameters for design use.	Expose temperature control model, preliminary design configuration(x) to critical environments. Measure temperatures at locations of particular interest and/or concern. a) Planetary orbit: Space temperature environment, simulated Mars intensity flight spacecraft thermal interface heat fluxes and simulated planetary thermal radiation. b) Post-separation: Simulated solar heat flux of Mars intensity and space environment.	a) Full-scale model*	Space chamber with Solar simulator, energy sink for space temperature simulation, energy sources for flight spacecraft thermal interface and planetary thermal radiation simulation. (e.g., ref. 5) Chamber capacity: 20-foot diameter (min) Vacuum required: 10^{-6} Torr (min) Solar beam diameter 16 feet (Mars intensity) Energy sources: heater panels Energy sink: cold wall Temperature measuring and recording instrumentation.
					b) Half-scale model* Materials preservation technique: $T_m/T_p = (L_p/L_m)^{1/3}$ T = Temperature, °R L = Length parameter m = model p = prototype	Same as above, except: Chamber capacity: 10-foot diameter (min) Solar beam diameter: 8 feet (Mars intensity scaled up to achieve similarity between model and prototype in regard to total absorbed thermal flux)

* Selection of the model scale to be made after closer definition of the hardware
(x)Resistance heaters simulate heat generating payload components

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3.0 STERILIZATION CANISTER SUBSYSTEM

3.1 REFERENCE DESIGN PERFORMANCE REQUIREMENTS

The sterilization canister subsystem encloses the entry vehicle and acts as a thermal control and sterilization barrier. As a thermal control barrier it is required to reduce the heat transmission to outer space, thereby reducing the power load required to maintain the flight capsule at a temperature sufficiently high to avoid degrading or destroying the mission objectives. As a sterilization barrier, it decreases the probability of recontamination of the entry vehicle after the sterilization process has been completed.

In addition to the above two requirements, the canister lid must be separated from the base and ejected to clear the subsequent path of the entry vehicle.

The reference canister design is a passive thermal barrier using surface emissivity values for control of the heat loss. The emissivity value for the exterior finish is required to be about 0.05. At the present time, a value has not been placed on the internal surface emissivity.

As a sterilization barrier the canister is required to maintain the 10^{-4} mission probability of transferring viable microorganisms from Earth to Mars. To maintain this probability, the canister is required to have the following characteristics: an internal pressure greater than that of ambient conditions (prior to Earth orbit) to assure an outflow of gases at all leaks; leakage holes smaller than the size of probable organisms (0.3 micron); design characteristics to prevent recontamination during pressure control and final lid opening.

In addition to the above requirements, the following must be accomplished.

1. The differential pressure must not exceed 1.0 lb/in.^{2} (For structural reasons).

2. Depressurization is required in Earth orbit so leakage conditions do not perturb the planetary vehicle.

3. Ejection velocity of the canister lid must be 1.5 ft/sec.

4. Lid separation must not degrade the operation of either the flight capsule or flight spacecraft by debris from the opening subsystem or by inducing uncontrollable perturbations to the system.

3.2 TECHNOLOGY DEVELOPMENT REQUIREMENTS

Technology development requirements of the sterilization canister result from the requirement for maintenance of the entry vehicle in a sterilized condition.

To maintain the sterile condition all entrances for viable microorganisms must be sealed and lid separation must be accomplished in such a way as to preclude vehicle recontamination. The critical items in technology development are the determination of means to detect migration of viable organisms, the determination of the capabilities of these organisms to migrate against a positive pressure gradient, and also the determination of methods to detect molecular leaks. These methods must be suitable for manufacturing purposes and not just useable in a laboratory. For instance, it will be difficult to locate a hole of the 0.2-micron size in a 184-inch diameter weldment. In addition, these leak detection methods must be developed for determining canister leaks prior to completion of the entire canister assembly.

Tests for the lid separation are mainly for sizing the charge and finding a method of containing the explosion products and debris. The technology is known, but the specific design is not. However, the method of measuring contamination by viable organisms when this separation system is activated presents a major problem and is discussed under recontamination in Section 3.4.

3.3 PRESSURE CONTROL AND SEALING INTEGRITY

Originally the intent was to keep the canister sealed and pressurized during the entire mission from the sterilization cycle to just prior to canister lid separation. Due to the perturbations that could be induced to the planetary vehicle in case of a leak developing in the canister, it has been decided to depressurize just after the launch phase of the mission. Thus, the canister leak rate serves as an indication of the hole sizes available for microorganisms to pass through, rather than for an indication of the supply of air required to keep the vehicle pressurized. The pressurization subsystem for this flight capsule reference design consists of the sealed canister, a fill valve, a relief valve, and a depressurization valve. Pressure regulation from the sterilization cycle to installation of the flight capsule on the flight spacecraft is done by ground handling equipment. The valves require individual element tests that involve helium leak detection techniques to evaluate sealing adequacy. These tests are straight-forward in nature and are not believed to present any unusual problems. However, to relate this leakage to hole size at these pressures and to find and evaluate these holes in the seals and welds requires development tests to determine a useable technique.

3.4 RECONTAMINATION

The sterilization system shall demonstrate, through an appropriate series of tests, its capability to provide the required degree of protection against microbial contamination.

Criteria and techniques must be established for evaluation of the performance of the sterilization canister system, after which the test program resolves to a determination of the maintenance of sterility, using various prototype and production units that will be exposed to mission environments.

The environments that are of primary concern are ground handling shocks, launch pad mating and ascent flight. In these environments the possibility exists that seals could momentarily open and allow passage of an organism, or that microscopic structural failure at joints and fittings could foster contamination.

One test will be the confirmation of the effectiveness of the sterilization cycle. The test article will consist of a full-flight capsule system consisting of parts, components, assemblies, structure, and subsystems that are identical to flight qualifiable equipment. The piece parts and subsystems shall be processed through the defined sterilization procedures and assembled in accordance with the manufacturing plan.

In order to ensure that the internal microbial burden has been completely killed, colonies of microbes, e.g., serratia marcescens, (S. M.)* would be established in locations within the flight capsule near points that are thermally remote. These colonies could be contained in closed capsules such that inadvertent capsule contamination would not occur. As the simulation of mission operation sequences is performed to demonstrate the adequacy of prototype hardware after exposure to all simulated mission environments, the planted capsules could be recovered and examined for effective kill provided by the sterilization system procedures.

The above test could be part of the thermal control tests; if not, the same canister could be used in the next test. The entry vehicle would not be necessary. The second test requires that the internal surface of the canister be assayed to assure noncontamination. A nutrient is then spread on the interior and the canister is sealed. The unit is then exposed to a sterilization cycle. After completion of the sterilization cycle, the canister is exposed to appropriate bacteria. This exposure is attained through the simulation of the vibration, shock and loading conditions of handling and launch, and the temperature gradients of space cruise. The unit is then thoroughly cleaned of exterior bacteria and assayed. The canister is then opened and the interior checked for contamination.

Recontamination during cruise can be caused by the entrance of microorganisms through the filter in the depressurization line. The effectiveness of this filter and of a labyrinth at the exit of the line should be the subject of an element test rather than a test with the canister assembly. The element test hardware would consist of the valve, filter and exit line attached to a container and placed in a vacuum chamber. With the vacuum pulled down to 10^{-6} lb/in.^{2} or less, microorganisms are introduced to the chamber. A time limit will have to be determined consonant with the density of microbes in the test chamber in relation to that assumed reaching the opening in space and matching the probability of entrance in these cases. At the completion of the exposure the test item would be withdrawn, the valve closed, and the exterior of the unit sterilized and cleaned. The container would then be opened and the inside cleaned with a sterile bacterial broth. The bacteria count is then taken from this broth.

To check for recontamination during opening of the canister lid, the procedure is as follows:

* Serratia marcescens is a form of aerobic bacteria, gram negative and non-spore forming which, when cultured, appear as visible colonies of a distinct red color, uncommon to other organisms likely to be found.

The sterilization canister's outside surfaces are painted with a powdered suspension of lyophilized (freeze dried) S. M. The canister containing the capsule is then placed in test chamber which has been examined for possible contamination by S. M. In a like manner, the capsule surfaces and the inner walls of the sterilization canister are assayed. If, by chance, S. M. is found present, other indicator organisms such as Bacillus globigii or Chromobacterim violaceum might be used. The test chamber is then pumped down to produce a vacuum. The canister is jettisoned and the capsule released. The capsule is so oriented that it will fall into a plastic bag after it is released. The bag is then sealed and removed from the test chamber. The bag's external surfaces are then washed down with disinfectant to prevent transfer or any S. M. to the capsule. The bag is carefully removed and the capsule exposed. The surfaces of the capsule are swabbed in square areas (6 by 6 inches approximately) and the swabs cultured in nutrient agar aerobically at room temperature for up to 72 hours. The absence of red colonies on the agar plates will indicate that the capsule was not contaminated by microorganisms from the surface of the sterilization canister.

3.5 CANISTER LID SEPARATION

The design for separation of the lid section of the canister consists of a mild detonating fuse (MDF) encased in a plastic case to retain the explosive residue. The detonation of the fuse expands the case against the canister shell forcing it to shear on the outer metal ring. The expansion also forces the lid away from the base.

The problems requiring development testing of the separation subsystem and resulting from the environments are the high sterilization temperature, the low temperatures during cruise through space, and the long exposure to high vacuum conditions during the cruise phase. In particular, degradation has been experienced in tests of commercial grade RDX after exposure to temperatures near 300°F for prolonged periods. Subsequent tests have proven most of this degradation can be avoided by improvements in the manufacturing and inspection techniques (tighter specifications). However, some degradation in the velocity of detonation is still unaccounted for and additional testing is required to assure that the degradation that occurs will not jeopardize the actuation of the separation system.

The cold temperature experienced during the cruise phase in space requires that tesing of explosive components to be accomplished to determined the changes in their actuation characteristics, and to determine the ability of the explosive to survive without chemical separation or mechanical cracking.

The high vacuum condition will be experienced by the separation system after the launch phase. The problems of high vacuum in relation to the separation subsystem will be the outgassing of explosive constituents and the possible cold welding of adjacent surfaces.

Other problems to be solved in the separation system development are to prevent hangup and/or excessive tipoff of separating parts. The canister separation subsystem has a separation plane oriented normal to the flight-axis direction. In separation systems of this type, the structures inherently provide substantial circumferential stiffening. The separated sections therefore have substantial hoop stiffness as contrasted with separation systems such as clamp-shell type shrouds that separate in longitudinal planes. For this reason dynamic flexure of the separated sections is of quite small amplitude and should not cause interference and collision between separated sections. The problem of hangup is, instead, primarily dependent upon the detonation of all MDF.

Tipoff can be induced by circumferential variations in elastic energy release due to nonuniform preload in mechanical multi-point tiedowns. In MDF-type joints tipoff is due to circumferential variation in explosive detonation velocity, core loading, and structural deformation caused by explosive forces. The logical development program is one that attempts to prevent the problems by extensive use of small sample tests. Comprehensive testing of representative sections of the preliminary design provides detailed guidance for design revision; involves the use of relatively inexpensive test samples; and allows acquisition of reliability information at an early stage in the program.

Table XXII presents the development tests that would be performed on components of this separation subsystem. The general purpose of the tests indicated is to evaluate the characteristics of the design, establish the performance of the explosive elements after exposure to degrading environments, and provide the necessary guidlines to evolve the preliminary design into a workable system.

Development and demonstration testing of the system will utilize a ballistic pendulum. Early development test samples would consist of a short cylindrical section with a full-scale diameter and would represent, in essential details, the preliminary design. The test sample would be mounted between two ballistic bifilar pendulums that are weighted to represent the respective masses of the spacecraft/capsule assembly and the lid section of the canister. Figure 2 shows the schematic design of the setup. Through proper calibration and measurement of pendulum displacements, and with equivalent mass moments of inertia of the pendulum, calculations can be made for gross impulse and tipoff forces (horizontal plane). The rotational and displacement measurements of the pendulums are made from high-speed motion picture analysis, light beam/photo sensitive paper, hot pen direct measurement or from break wires. This techniques has been used extensively and successfully in spite of the lack of 6-degrees of freedom.

The lack of complete freedom in a zero-g field can be compensated in the ballistic pendulum technique by constructing and orienting the test specimen to have the highest explosive energy gradient (or other impulse producing source) in the horizontal plane. Thus with the explosive core load distribution as shown in Figure 2, the maximum tipoff forces are determined.

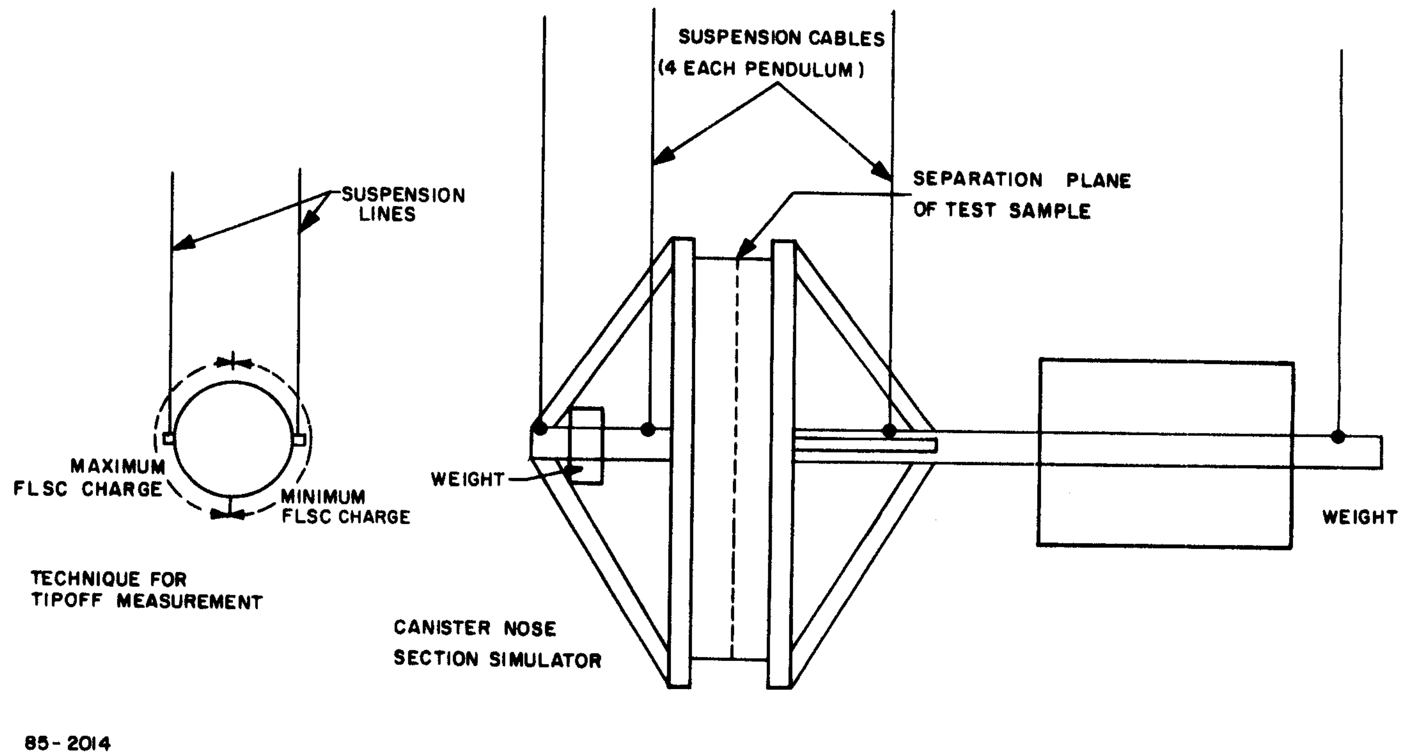

Figure 2 SCHEMATIC TEST SETUP FOR SEPARATION TEST USING BALLISTIC PENDULUM

TABLE XXII

CANISTER SEPARATION DEVELOPMENT TESTS

A. System Function	B. Component	C. Test Objectives	D.
1. Canister Lid Separation	a) Mild Detonating Fuze	Measure velocity of detonation and plastic case characteristics with candidate filers, before and after exposure to sterilization temperatures and at low temperature and vacuum simulation of space. Sterilization Test Temperature: 293°F (145°C) three cycles of soak at 36 hours each. Space Test Vacuum: 10-6 torr Space Test Temperature: -150°F (-71°C)	To be determined
	b) Detonators	Measure output degradation from sterilization and space environment. (see la for temperature/vacuum range).	Dent Test Mil. Std. 316 Pressure Bomb Tests
	c) MDF/Detonator	Demonstrate detonation of MDE by selected detonator and determine gap and offset allowables.	50 ignition tests with flat samples.
	d) Flat sample separation joint with MDF	Establish backblast contaminant, separation impulse, local structural deformation and fastener requirements.	20 tests with flat separation joint samples.

The accuracy of impulse measurement and tipoff determination is a function of the length of the ballistic pendulum suspension lines. Environmental chambers do not generally allow use of long suspension lines (order of 25 feet). Tests are usually run in a normal atmospheric environment using long suspension lines to establish impulse and tipoff, and then confirmatory tests (with shorter lines) are run in vacuum environments to demonstrate performance of explosive trains, measure gas-shock pressures, and examine particle backblast containment. (Large chambers are available - Langley 55-foot - but the extensive testing required may cause schedule problems.)

The backblast containment in early development hardware is determined by use of carbon blacked or painted witness plates positioned inside the test sample in proximity to the separation joint. Examination of the plates, after test, locates areas of design deficiency (usually around detonators) and, from penetration measurements, allows judgement of the severity of particle backblast. This same technique, but with witness plates externally placed, shows particle dispersion caused by the exploding MDF and allows evaluation of the possible impact on other parts of the system.

Bikini gages, (plastic membranes of different diameter calibrated to break at various pressures) are also used to evaluate performance.

Demonstration tests of the fully developed sterilization canister would be performed using the same test techniques but would utilize full structures representative of the final design (prototype and flight acceptable). In addition, a full operating flight capsule would be used in at least one test to demonstrate particle backblast effects on thermal control coating, structural stress wave-shock and gas-shock effects on electronics (relays are of particular concern), and confirmatory measurement of gross impulse. These full-scale separation tests would be performed in conjunction with or as part of the tests recommended in Section 3.4.

4.0 SEPARATION SUBSYSTEMS

4.1 FLIGHT SPACECRAFT - ENTRY VEHICLE SEPARATION SUBSYSTEM

4.1.1 Reference Design Performance Requirements

The entry vehicle shall be separated from the flight spacecraft (at the forward ring of the flight capsule adapter section) by a V-type clamp-ring. This clamp-ring shall be released, upon signal, by the actuation of any one of four explosive nuts. The separating force is provided by 10 compression springs, two of which are used to overcome the friction of the two electrical connectors; while eight provide the impulse to obtain the separating velocity. The requirements of the system are as follows:

1. A minimum separation velocity of 1.5 ft/sec.

2. A clean separation with all parts retained on the adapter if possible and no debris ejected that might degrade the operation of entry vehicle or flight spacecraft equipment.

3. Isolation of shock due to explosive actuators so as not to damage electronic equipment.

4. Relative tipoff rates of less than 3 deg/sec to avoid interference between vehicles as the entry vehicle is ejected.

5. Minimum perturbation to the spacecraft so as not to disturb Sun-Canopus orientation.

4.1.2 Technology Development Requirements

The separation subsystem used for separating the entry vehicle from the spacecraft is a standard design used in many other types of equipment; however, the environments in which it functions are generally more severe than previously encountered. The problems associated with the environments are the temperature range, from 275°F (135°C) during the sterilization cycles to approximately - 100°F during planetary transfer and the storage in high vacuum during this transfer. The temperature range presents a problem mainly to the explosives where degradation in the detonation rate has been experienced in commercial grade explosives exposed to temperatures in the sterilization range. Preliminary tests indicate that more stringent specifications, especially in the area of impurities, will solve this problem. Some devices have already been exposed to these temperature extremes and have been actuated. If hermetically sealed cartridges are used to avoid the vacuum problem, the tests required would be explosive charge sizing tests and environmental tests to check the integrity of the design.

The space vacuum (estimated at around 10^{-16} torr) presents a problem to parts of the separation assembly that are required to move relative to each other. As stated, the explosives, if hermetically sealed within their own containers, will probably not be affected; however, the V-type clamp presents a large surface contact area that must not bond to the rings being clamped. The vacuum conditions, by removing the surface films from the adjacent parts, increases the probability of this bonding action taking place. One solution is to coat the surfaces with a very slow-to-outgas material that doesn't have the strength to bond the adjacent parts together (such as silicone grease) or with Teflon. The system tests of prime concern, are to confirm the design by testing the performance of the assembly to assure it meets the requirements of section 4. 1. 1.

4. 1. 3 Explosive Nut with Bolt Ejection

4. 1. 3. 1 Test Objectives and Description

The objective of this test is to check the function of the explosive nut including retention of explosive products and to evaluate degradation due to sterilization and space storage temperatures.

4. 1. 3. 2 Test Facilities and Equipment

The following facilities are needed:

1. Oven -- To heat test sample to 300°F.

2. Vacuum Chamber -- To produce 10^{-6} lb/in.^{2} pressures and -100°F temperature.

3. High Speed Cameras -- With greater than 4000 frames/sec to observe actuation.

4. Accelerometers, Strain Gages, Pressure Gages and Recording Equipment -- to measure effects of explosion.

4. 1. 3. 3 Test Conditions

Since this test is to determine performance degradation of the explosives, samples must be tested at room temperature and compared with samples subjected to the mission environments and then tested. These environments are:

1. Sterilization Heat Cycle -- in accordance with JPL specification Vol-50503-ETS.

2. Launch Vibration -- Inputs to agree with this report Volume III, Book 2, Section 4. 3.

3. Minimum Temperature -- -100°F

4.1.4 Entry Vehicle Subsystem Separation

4.1.4.1 Test Objectives and Description

The test techniques for development and demonstration testing of this subsystem will be performed in a manner analogous to the test of the sterilization canister lid separation (Section 3.5).

Initial testing will utilize full-diameter circular sections suspended from ballistic pendula and final demonstration tests will utilize fully representative hardware.

There are two essential differences to be observed between the application of this test technique to the canister lid separation and the entry vehicle separation. The first difference is that the structural shock load for release of clamps and bolts will probably be less than the shock loads from the MDF separation and gas-wave shock and debris will not be present. Therefore, it is not as necessary to have tests to evaluate shock effects of this system on the electronic packages.

The second difference deals with the problem of misalignment between the separating pendula which can cause binding of spring guides or intermeshing parts in the area of the separation planes. This problem can be countered by adjusting the pendulum masses such that their displacement with time from the initial position is the same (equal initial velocity) with the result that no misalignment will occur. Also, the length of suspension lines must be sufficient to allow the separating impulse to cause complete extension of separating springs and full extraction of intermeshing parts. With these differences noted, the test technique is equally applicable to the entry vehicle tests.

4.1.4.2 Special Test Facilities and Equipment

This test will utilize the same equipment as required for the sterilization canister lid separation test described in section 3.5.

4.1.4.3 Test Conditions

Ambient test conditions should be used.

4.2 ENTRY SHELL - SUSPENDED CAPSULE SEPARATION SUBSYSTEM

4.2.1 Reference Design Performance Requirements

The entry shell shall be disconnected from the suspended capsule by a V-type clamp ring. This clamp ring shall be released, upon signal, by the actuation of any one of four explosive nuts. The separating force is provided by the drag of the deployed parachute system and the separation of the two structures separates the electrical umbilical. The requirements of the system are as follows:

1. A clean separation with all parts retained on the entry shell and no debris ejected that might degrade the operation of the suspended capsule equipment.

2. Isolation of shock due to explosive actuator so as not to damage electronic equipment.

3. The entry shell must not bump the suspended capsule during separation.

4.2.2 Technology Development Requirements

The technology development requirements are the same as those required under the flight spacecraft-entry vehicle separation subsystem (Section 4.1.2) except that tests to confirm the performance of the assembly require release of the entry shell under simulated worst-case parachute loading conditions, and pendulum tests are not required.

4.2.3 Explosive Nut with Bolt Ejection

To be done under 4.1.3.

4.2.4 Entry Shell Subsystem Separation

4.2.4.1 Test Objectives and Description

Entry shell - The separation of the entry shell (heat shield and structure) from the suspended capsule will occur in the supersonic regime of entry flight. The vehicle will be at near-zero angle of attack. The main parachute system will be deployed at these flight conditions to further decelerate the flight capsule.

At some time shortly after peak parachute loading, the separation system will release the entry shell. The difference between ballistic coefficients of the shell and parachute/suspended capsule configuration will cause the shell to fall away from the payload.

To assure the clamp ring will not cause a temporary hangup condition that will in turn cause bumping between the entry shell and the suspended assembly, a test should be run with the vehicle in a vertical nose-down position and hung at a height above the floor. If future aerodynamic studies show a spin is induced on entry, an initial spin could be added to the test vehicle. Release actuation characteristics would be determined by linear extensiometers and recorders and by high speed motion picture coverage of the test setup as the clamp ring is released.

Another test will be run simulating the dynamic conditions of the separation. These conditions will be obtained by dropping a simulated entry vehicle from a tower to achieve the required initial loading. A restraining cable connected to the parachute swivel will simulate the parachute drag load after extension of the cable. The force history of the parachute drag will be approximated by use of a series of springs and dash pots to restrain the drop cable. Data acquisition will be by means of high-speed cameras, accelerometers, and recording equipment.

4.2.4.2 Special Test Facilities and Equipment

1. High Speed Cameras -- 1000 frames/sec.

2. Recording Equipment

3. Accelerometers -- To measure 20-g parachute load and show bumping conditions.

4. Tower -- in excess of 100 feet

4.2.4.3 Test Conditions

1. Environment -- Still air.

2. Loading -- 20 g

3. Angle of Loading -- Variable.

5.0 PARACHUTE SUBSYSTEM

5.1 REFERENCE DESIGN PERFORMANCE REQUIREMENTS

The two major items to be investigated via ground testing are (1) The flow field behind the blunted cone throughout the Mach number range of interest, and (2) the performance characteristics of the parachute itself including aerodynamic coefficients, inflation, stability and shock-load attenuation.

Other ground testing, including initiation devices and/or circuitry and deployment mechanisms is standard and will not be discussed herein. Note that all of the test components must be put through the sterilization criterion before commencement of testing.

5.2 BLUNT-BODY WIND-TUNNEL TESTS

Flow-field characteristics behind the blunted cone are required and can be accomplished in the wind tunnel. Results from Mach 0.1 to 1.2 are required across the entire traverse of the tunnel so that, q/q_{stag} and p/p_{stag} can be measured at varying distances behind the forebody stagnation point. The results of these tests, will indicate whether or not inflation of the chute is choked due to the blunt-body flow-field effects.

5.3 PARACHUTE WIND-TUNNEL TESTS

The performance characteristics of the parachute, both at deployment and during its subsonic descent, can be established via wind-tunnel testing. The parameters to be established are (a) drag coefficient, (b) stability (aero coefficients), (c) opening shock-load attenuation, (d) inflation, (e) canopy porosity effects and (f) blunt-body wake effects. Wind-tunnel instrumentation to evaluate the above parameters is standard in nature and will not be discussed here. Note that the results of the test conducted will be limited by scaling uncertainties.

6.0 PROPULSION SUBSYSTEM

6.1 REFERENCE DESIGN PERFORMANCE REQUIREMENT

The separated vehicle requires a ΔV capability of 1400 ft/sec with a single firing cycle. In addition only subsystems that would have their state-of-the-art established by September 1966 should be considered for use in the flight capsule design. The other requirements were: sterilizability, reliability, space storageability, total impulse accuracy and 101,600 lb-sec total impulse.

6.2 TECHNOLOGY DEVELOPMENT REQUIREMENTS

The requirement that the rocket motor must meet its operational performance after being subjected to sterilization and long term space storage imposes a condition to which space motors under development had not been previously subjected. Sufficient testing has been accomplished, however, to indicate that new technology is not required.

6.3 STERILIZABLE PROPELLANTS

Sterilizable propellant development has been underway for over two years by Thiokol Chemical Corporation. The development effort during the period has been with TP-H-3105 propellant, and this is the propellant being used in the reference propulsion system. The most recent work is a program Thiokol has under contract with JPL for a "Design Study of Heat Sterilizable Solid Rocket Motors." Test results to date indicate that TP-H-3105 propellant is able to meet the sterilization requirements without degradation in motor performance.

The details (rather than the basic nature) of the sterilization requirements and procedures are changing continuously and probably will continue to do so over the next two years. Therefore, it is the considered opinion of those concerned with this area of development that TP-H-3105 is a satisfactory propellant and that no extensive development program is required to obtain a sterilizable propellant. Development efforts similar to those underway will continue primarily to evaluate the limits of the propellant under various environments rather than to determine basic design information.

The propellant development portion of this rocket motor development will follow the approach used for an existing propellant, but being tailored to a specific motor design. In addition to the sterilization environment the motor will be exposed to a space environment of 10^{-6} Torr for up to one year necessitating some propellant space aging tests. Some work has been done with similar propellants and the results have indicated that no critical problems will be encountered, but, because the time period for this application is longer than those previously planned, more testing over a longer period will be conducted. The testing referred to was with a complete Surveyer ignition system, consisting of the safing and arming device, squib and the pyrogen. The units were put under vacuum of 10^{-5} to 10^{-6} torr and cycled several times between 0° and 100°F. The units operated satisfactorily; the performance was statistically indistinguishable from tests at sea level. This same basic ignition system design concept is used for the reference propulsion system.

In a very extensive space aging program, Thiokol investigated the space storability of several propellants at -65°F, and 175°F, at both atmospheric pressure and under a vacuum of 10^{-6} to 10^{-7} torr, for time periods of up to eight weeks. One of the propellants, TP-H-1050, is of the same basic propellant family as TP-H-3105. After eight weeks under high vacuum at ambient temperature the modulus had increased somewhat, as had the burning rate, the ease of ignition and impact sensitivity. These changes were not considered serious; however, considerable reduction in their magnitude was found possible by modifying the polymer from which the propellant was made.

6.4 MOTOR DEVELOPMENT

The motor development program required would follow the same approach as is used to develop similar rocket motors for space applications. Because of the sterilization requirement, additional tests would be required to verify, that there are no degradation effects due to the sterilization procedures and environments. The development program would also have a phase to determine and evaluate special manufacturing and assembly techniques because of the sterilization requirement.

The one area where there is little preliminary data on the effects of sterilization environments is ignitors. The ignitors use the same propellant as the motor, so that no difficulty is anticipated with the ignitor propellant. The safing and arming and squibs have not been exposed to heat sterilization environments and are considered an unknown. Some work has been done to determine the squib designs that are compatible with the sterilization environment with favorable results. No work to date has been done to determine the RFI limits of sterilizable squibs. It is felt that state-of-the-art development is not involved.

The development program will be followed by a design verification and a type approval test program. Again, this would follow the presently established programs with the exception that sterilization and space aging impose additional tests.

6.5 ASSEMBLY AND HANDLING

There will be two groups of materials flowing into the propellant manufacturing and motor assembly area. The first group consists of the non-sterile raw materials and components which must be thermally sterilized, and the second group consists of the sterile and sterilely packaged raw materials and components for which only surface sterilization is required. Since the propellant is to be manufactured and the motor is to be assembled under clean room constraints, the thermal sterilization or the ethylene oxide/freon surface sterilization must be accomplished, as applicable, on each of the raw materials and components before it enters the clean processing and assembly area. The clean-room facilities will be for mixing, casting, finishing and assembly operations. The clean room will adhere to the specification of "Class 100, 000" of Federal Standard 209, which is the equivalent of the Air Force "Standard Clean Room (Operational)". The clean room itself will be divided into several modules for safety reasons. Modules will be for:

a. Shipping and receiving

b. Assembly of motor case and of casting hardware before casting

c. Mixing, deaeration, and casting of the propellant

d. Cutback and finishing of the cured grain

e. Final motor assembly after cure, i.e., addition of ignition system.

While the function of all modules is self-evident the shipping and receiving module requires some additional explanation. This module would have separate access (for materials only) to an uncontrolled area outside, in addition to normal, unrestricted intercommunication with the other clean room modules. Communication between the shipping and receiving module and the uncontrolled area will be through three interchanges: (1) an oven with the door opening into the clean area and one into the uncontrolled area, (2) an ethylene oxide/freon decontamination chamber, likewise with the two doors as above, and (3) a clean interchange, again with two doors.

The flow of all materials into the clean room is through this shipping receiving module. Items requiring thermal sterilization, ammonium perchlorate, etc., will be placed in the oven from the uncontrolled side, thermally sterilized and removed from the clean side. Items requiring chemical sterilization, polymer drums motor cases, etc., similarly will pass through the ethylene oxide decontamination chamber. All items placed in the oven or the decontamination interchange from the uncontrolled area will be normally clean.

Likewise, items to be shipped out of the clean room will go through the clean interchange of this module. This finished motor will be sterilely packaged before entering the interchange.

Within the clean room, the usual propellant manufacturing processes will be employed. The motor will be assembled and the propellant deaerated and cast. For curing, the motor will be packaged in a sterile envelope and removed from the clean area to the regular curing ovens. After cure it will be returned to the clean room for cutback and finishing. At this time the ignition system will also be added. The final steps in the motor production task will be the sterile packaging and sealing of the motor so that it can be removed from the clean area for crating in preparation for shipment.

The handling procedures for the motor after it has been packaged for shipment will be similar to those now established for existing motors in this size class. The procedures will be modified to be compatable with the established sterilization requirements and procedures. At the time the motor is ready for shipment, the motor manufacturer will issue a document to cover the handling, checkout, and installation procedures to be followed in the field.

7.0 ATTITUDE CONTROL SUBSYSTEM

7.1 REFERENCE DESIGN PERFORMANCE REQUIREMENTS

7.1.1 Inertial Reference Subsystem (IRS)

The inertial reference system consists of a four gimbal inertial platform, a digital computer and a three-axis sensor system for rate limiting. The four gimbals on the platform are required to permit full flexibility and angular freedom. The platform also contains the accelerometers for data purposes, thus providing inertially referenced acceleration data. The digital computer provides the means of transforming the platform gimbal angles into the proper reference frame for commanding the vehicle reaction control system. The computer will accept angular commands from the CC & S and provide the logic to control the reaction control valves. The computer will perform the proper transformations from the IRS gimbal angles to command the TV camera gimbals along the local vertical. The computer has the ability to provide an integration of the accelerometer data for providing velocity information if desired.

Control Logic

The control "deadband" shall be ± 0.5 degree from the nominal commanded angle for the yaw and pitch axis and ± 1.0 degree for the roll axis with a hysteresis factor of 10 percent.

Error Sources

Error contributions (1 sigma) in the computation and control of the angles are:

G-Insensitive Gyro Drift	0.4 deg/hr
G-Sensitive Gyro Drift	0.3 deg/hr/G
Gimbal Readout Error	0.1 degree
Computation Error	0.1 degree

7.1.2 Cold-Gas Reaction Control Subsystem

The reaction control subsystem proposed for the nullification of separation rates, orientation, roll control during de-orbit thrusting, reorientation and limit cycling is basically a cold-gas system utilizing gaseous nitrogen as the propellant. This cold-gas system consists of two identical subsystems each utilizing a central regulator, pressure-vessel, vent and fill manifold, squib valve, filters and solenoid valve nozzle assemblies. The twelve solenoid valve nozzle packs provide the required three-axis control in couples. This redundant system approach is utilized to provide the necessary mission safety margin in the event of failure modes such as:

a. developing a gas leak in one of the subsystems, and

b. a valve failing to close.

In the case of one of the failures described, one nozzle of the couple will be lost and the other nozzle will allow completion of the mission.

Thrust Levels per Axis

Yaw	Two thrustors at 0.5 pound and = 1.0 pound (1 pound at 7.33 feet = 7.33 ft-lbs.)
Pitch	Two thrustors at 0.5 pound and = 1.0 pound (1 pound at 7.33 feet = 7.33 ft-lbs.)
Roll	Two thrustors at 0.5 pound and = 1.0 pound (1 pound at 7.33 feet = 7.33 ft-lbs.)

Total Impulse

Required	75 lb-sec
Stored	248 lb-sec

Response Parameters

Time Delay	0.020 second
Time Constant	0.005 second

7.1.3 Hot-Gas TVC Subsystem

A hot-gas reaction subsystem is proposed for maintaining the desired vehicle attitude during the de-orbit thrust application. This system is composed of four identical solid propellant gas generator packs, each consisting of a solid propellant gas generator, closely coupled to two solenoid operated hot-gas valves and two reaction nozzles. These packs are located on the vehicle as couples on both the pitch and yaw axis, to realize an increase of mission safety margin in case of a generator or valve failure.

Thrust Levels per Axis

Yaw	Two thrustors at 25 pounds and 50 pounds (50 pounds at 7.33 feet = 366.5 ft-lb.)
Pitch	Two thrustors at 25 pounds and 50 pounds (50 pounds at 7.33 feet = 366.5 ft-lbs)

Gas Generator Performance

Required	1225 lb-sec
Stored	3500 lb-sec
Specific Impulse	180 seconds
Burn Time	35 seconds

Response Parameters

Time Delay	0.020 second
Time Constant	0.005 second

Temperature Limits

Operation	-100°F to 140°F
Storage	-100°F to 140°F
Sterilization	300°F maximum

7.2 TECHNOLOGY DEVELOPMENT REQUIREMENTS

7.2.1 Inertial Reference Subsystem

The inertial reference subsystem design represents currently available hardware designed for missile applications. The components are therefore capable of withstanding the normal missile launch environments. However, the mission requirements for this program present several factors that require investigation. These factors include the sterilization cycle; the long vacuum soak; and the low-temperature soak.

The manufacturers of the IRS components and subsystems indicate that in general, with, minor redesign, the currently available components will be able to perform within acceptable limits despite the harsh operating environments and sterilization cycle. Some of the components of the IRS (gyros, electronic computers, etc.) have been successfully used on space programs such as Ranger and Mariner. The general conclusion can be drawn that only normal development will be required.

7.2.2 Cold-Gas System

The cold-gas reaction control concept has been used extensively for the attitude control of ballistic missiles, satellites and interplanetary vehicles. Numerous components of various sizes have been flight proven during programs such as LORV, Mariner Ranger, RMV, OSO and OAO, to cite a few. Component design for space flight-worthy equipment, such as pressure vessels, regulators, and solenoid and squib valves, has become a highly developed technology in the past few years. Further development of these components is required only for the requirements imposed by missions of a highly specialized nature; Sterilization is one such requirement.

Sterilization subjects the system component to elevated temperature, (300°F) for a period of approximately 36 hours. Therefore, special design techniques must be employed to maintain dimensional stability in the parts for regulators and solenoid valves, since their performance is dependent on close fitting sliding members. Furthermore, the parts must maintain their dimensional stability when subjected to long periods in the space environment (-100° to 140°F)

Squib initiators have been developed with functional capabilities between temperature limits of -300°F and 450°F. Hence, squib compatibility with the environments does not present significant problems.

The design of the pressure vessels and line complexes is straight forward and major problems are not anticipated.

System assembly and performance tests under ultra clean sterile conditions necessitates the exercise of strict personnel and facilities environmental control. Formulation of procedures will consist basically of a modification of those presently established to ensure compatibility with the requirements.

7.2.3 TVC Hot-Gas System Technology Status

The components of the TVC hot-gas system are representative of state-of-the-art hardware employed in ballistic missile and manned aircraft systems. Solid propellant gas generator technology is presently based on proven design techniques successfully applied to such representative systems as SUBROC, POLARIS, MINUTEMAN, etc. The reference design selected for the solenoid valve is the Minuteman hot-gas roll-control system valve design. In addition to being a fully flight qualified configuration, the subject valve incorporates configuration features which are attractive for deep-space applications.

The major portion of the development program required for the TVC is directed at insuring compatibility with the sterilization and long term space mission environments.

Gas generator performance is readily predicted for the type of device common to present weapon system designs. The advancement of design technology demanded by the TVC gas generator implies determination of propellant and ignitor performance characteristics exhibited during and after exposure to environments of unusual severity. The subject environments represent an extension of those presently encountered on typical missile systems. Ability to meet the sterilization requirement of 36 hours at 300°F will require further testing. The autoignition test run on current solid propellant attitude control rockets of 200°F for 6 hours is an indication of the ability of existing designs to meet elevated temperatur requirements. Furthermore, the need to operate in a -100°F thermal environment should not present difficulty in view of test data accumulated demonstrating -65°F as an acceptable temperature level for solid propellant rockets and jet engine starter cartridges.

Adequate performance data are available on the reference design solenoid valve to verify the functional integrity of the device as a hot-gas valve. The problems associated with utilization of the valve in an extended duration space mission should not require configuration revisions of any magnitude.

The lack of closely fitted precision parts and metallic surfaces in sliding contact characterize a proven component as highly attractive for development to withstand the extreme temperature range associated with the TVC system.

7.3 INERTIAL REFERENCE SYSTEM DEVELOPMENT TEST PLAN

The IRS has three major assemblies: the inertial platform; the computer; and the rate gyro assembly. Although each is a separate assembly, they have common operating requirements, and a general development test plan is applicable to all of these assemblies.

Each of the assemblies (platform, computer, and gyro package) will be tested on the assembly level to validate the predicated performance and to ensure acceptable performance during or after exposure to the applicable environmental conditions. In the case of the platform, this will include verifying the drift (G, G^2 and non-G sensitive) characteristics of the platform, its readout capability, and dynamic response characteristics. The computer characteristics will be checked by employing a checkout program for the computer that will exercise all of the computer circuits. The computer program operating characteristics will be determined by checking expected output against actual output. This technique is employed in the checkout of large digital computers. The gyro package check will verify gyro signal output as a function of input rate and the dynamic response characteristics of the gyro package.

All of the assemblies will be checked before and after exposure to a normal sterilization cycle. The tests will be the normal acceptance test procedures for verifying performance to the applicable performance specifications. The same testing will be required for exposure to the powered flight environments. Since the system is not required to function during this phase, only "before" and "after" testing will be performed.

The testing of the assemblies while subjected to outer space environments, (cold soak and vacuum) becomes more involved in that performance must be verified while subjected to these conditions. The same acceptance test techniques will be applicable with the stipulations that they be performed by remote control. Another factor which must be considered is the time duration of these tests. The normal exposure will be in excess of 6 months, and consideration must be given to proving acceptable performance after long term exposure. This will undoubtedly mean a special facility dedicated solely to long term vacuum and temperature exposure investigations for all systems. However, steady-state conditions can be met in far less time to prove short term effects which will prove the majority of the design goals.

Aside from the individual assembly tests stated above, it will be necessary to repeat the testing in an assembled subsystem configuration to provide proof of the subsystem compatability and performance for integration into the vehicle system. The IRS does have an additional test requirement peculiar to this system. Since the IRS must be active during the entry phase to provide stabilization data after parachute deployment, it will be necessary to subject the IRS to the expected entry angular dynamics to verify platform performance. This will be performed on a three-axis servo-driven flight table by mounting the IRS platform to the inner table gimbal and programming the table through the expected entry angular rate time history.

The facility requirements for the IRS testing will include those required by most systems (vacuum chambers, , temperature chambers, shake tables, etc.) and several generally available minor facilities such as precision indexing tables, autocollimators, isolation pads, and theodolites. The only major facility requirement is a three-axis flight table.

7.4 COLD-GAS SYSTEM DEVELOPMENT TEST PLAN

The following test plan has been devised to outline the procedure used to corroborate theoretical calculations of performance and structural integrity and to insure system compatibility with the mission requirements. All components will be subjected to the sterilization cycle and its effects will be evaluated. Long term cold-soak tests (-150° F) will be used to evaluate material structural behavior. The effectiveness of the sterilization itself will be evaluated in another phase of the program. Assembly of components and systems and in some cases tests, will be conducted in ultra clean rooms.

7.4.1 Pressure Vessel

Pressure vessel design calculations will be corroborated with laboratory burst tests. Adjustments will be made to the burst test pressures to account for the increase in vessel internal pressure and decrease in material strength when the vessel is subjected to the elevated sterilization temperatures. Areas of stress concentration and general stress levels will be investigated with stress-coat and strain-gage techniques, respectively. Environmental tests simulating space and Mars entry flight conditions will be conducted to investigate stresses due to cyclic temperature changes and vibrations and material structural integrity at low temperatures (-140° F). Pressure vessel fill tests will be conducted to determine the fill rates and cutoff pressures.

7.4.2 Pressure Regulator

Bench tests will be conducted to establish a reference performance level. These bench tests will investigate parameters such as response, repeatability, resetability, resetability, errors, regulation, and leakage. Components will be subjected to sterilization environments after which performance bench tests will be conducted to determine the effects of the sterilization temperatures and fluids. Compatibility of lubricants with deep-space environments will be determined.

7.4.3 Nozzle Solenoid Valve

Tests will be formulated to determine the compatibility of this component with the high temperature soak experienced during sterilization. Bench tests will be conducted to determine valve response characteristics, equivalent orifice size and leakage. These parameters will be checked before and after valve sterilization. The nozzle solenoid valve assembly will be evaluated for thrust transient response, steady-state accuracy and repeatability.

7.4.4 Squib Valve

Tests will be devised and a test matrix formulated to establish the squib valve fire reliability. A number of specimens will be environmentally tested and fired for verification of electrical characteristics. Flow checks will be made to evaluate the valve orifice size after firing.

7.4.5 System Development

Breadboard and prototype hardware will be fabricated to: a) check system performance, b) establish pressure drops, c) establish build up procedures, d) determine flow rates and tolerances, e) evolve cleaning procedures, and f) establish compatibility with vehicle installation.

The system will be reviewed to assure compatibility with vehicle design as related to the placement of fluid lines, tiedowns, joints, support brackets, leak detection, and general access to the componentry.

The minimum total impulse validation test will be conducted on a breadboard system. Pressure drop calculations will be verified for both the steady flow and the nozzle cross-coupling condition.

7.5 TVC SYSTEM DEVELOPMENT TEST PLAN

The following development tests are necessary to verify the theoretical design techniques incorporated in prototype hardware.

7.5.1 Gas Generator

The purpose of testing the generator on the component level is to validate predicted theoretical performance characteristics, demonstrate lack of susceptibility to system environments. and to provide parametric information required for successful total system integration.

7.5.1.1 Performance Tests

The operation of the generator will be verified for function time, ouptut flow characteristics, ignition response, operating temperature, erosion of the metering orifice, post-firing initiator characteristics, working and ultimate stress levels in the housing, external skin thermal gradients, mounting structure compliance, and deliverable impulse. Performance testing will be used to establish a design configuration. Included in these first tests will be checks of mass parameters for verification of preliminary information provided for vehicle design.

7.5.1.2 Sterilization Test

The objective of this test is to ensure that the sterilization process does not impair generator performance and that the allowable microbe count is achieved. The tests required for insurance of insensitivity of performance to the process would reflect the standard "before and after" checks typically associated with an environmental exposure. The initiator evaluation would include testing of electrical parameters such as no-fire current, static discharge sensitivity, and bomb calorimeter evaluation of caloric output and sure-fire characteristics. Evaluation of total generator susceptibility would require repeating those tests listed under performance tests which would indicate any variation in the thermo-chemical properties of the propellant. Evaluation of conformance to the allowable microbe count level will necessitate extensive dissection and examination of all generator elements.

7.5.1.3 Environmental Tests

The purpose of conducting environmental tests is to establish insensitivity of the generator design to the conditions of booster powered flight and prolonged space flight. As a "one-shot" device, the generator test specimens will be tested according to a predetermined test matrix. Firing each generator after a specific sequence of environments will yield performance data necessary to define the demonstrated reliability of the item. Firing a generator will involve performance monitoring as described under performance tests. The array of environments will simulate the mission environments within the limitations of test equipment and program schedule commitments. The simulation of deep-space hard vacuum with radiation and temperature extremes is representative of a condition wherein limitations of test equipment may be a factor. The simulation of vacuum soak periods of up to one year will undoubtedly be affected by program schedule commitments. Test equipment required will be associated with simulation of combined space vacuum, radiation, and temperature.

7.5.2 Solenoid Nozzle Valve

The objective in testing the solenoid nozzle valve on the component level is to demonstrate compliance with design objectives with a simulated flow provided by a laboratory facility hot-gas source. The verification of insensitivity to mission environments and evaluation of actual performance criteria is thereby achieved; valve design compatibility with the remainder of the system is reserved for system tests.

7.5.2.1 Performance Tests

The purpose of performance tests is to measure compliance of the hardware to theoretical design calculations. Testing will be conducted with simulated gas generator flow provided by a laboratory hot-gas supply supplemented with selective tests using cold gas. Parameters to be evaluated would include: nozzle thrust (both in atmosphere and in a hard vacuum), nozzle efficiency, thrust vector pointing accuracy, mounting structure compliance, exterior skin temperature monitoring, erosion of the nozzle throat section, evaluation of working and ultimate stress levels, leakage rates of the closed position valve mode, electrical power demand, transient response characteristics, measurement of valve equivalent sharp edged orifice size, repeatability, and frequency response performance. Test data will be used to corroborate preliminary design estimates of dynamic and electrical performance criteria utilized in the total TVC system synthesis.

Test equipment would include a laboratory hot-gas supply, hot-gas flow stream instrumentation, and a thrust vector evaluation test stand.

7.5.2.2 Sterilization Tests

The solenoid valve requires evaluation for sterilization feasibility to satisfy the identical objectives cited earlier for the gas generator. Testing would concentrate on those areas likely to be susceptible to a prolonged elevated temperature soak such as material properties, electrical and magnetic circuit parameters, and mechanical distortion. Testing of conformance to the standard of sterilization would require the same procedures under the gas generator.

7.5.2.3 Environmental Tests

The objective of environmental testing is to ensure compatibility of the valve configuration with the mission environments. Performance will be monitored before and after each simulated environment to detect any design weaknesses. Representative features to be checked would include distortion of parts due to thermal cycling and degradation of the solenoid actuation force through faults introduced in the electro-magnetic circuit.

7.5.3 Tubing Complex

The objective in testing the tubing complex is to verify theoretical design calculations and to determine the actual performance characteristics necessary in appraising the compatibility of the gas generator and solenoid nozzle valves.

7.5.3.1 Performance Tests

Performance testing will concentrate on examining the structural adequacy of the design and the variation of flow-stream properties caused by tubing parameters. Preliminary tests employing a cold-gas simulated generator flow will be complemented with use of a laboratory hot-gas supply for subsequent tests. Testing will be used to evaluate the following principal features: pressure drop, working and ultimate stress levels, external skin temperatures, cooling of the flow stream, thermal expansion. and suitability of the materials with the thermal shock associated with introduction of a hot gas to a -100° F stabilized tubing complex.

Evaluation of the cross coupling of solenoid valves operating with a tubing complex and simulated gas generator flow provides performance data essential to prediction of the actual TVC system dynamic operating characteristics. The variation in opposed nozzle thrust values induced by simulated mission duty cycles imposed on the valves will be determined and employed to predict actual system performance.

7.5.4 TVC System Tests

Development testing of the TVC system is for the purpose of determining compatibility of the system components in performing according to the design objectives. An additional goal is to substantiate insensitivity to mission environments and to establish the desired confidence in the conceptual integrity of the system necessary for final drawing release. Performance tests will be composed of a series of measurements essentially repeating tests run on a component level for each system element. Evaluating all performance characteristics on a system level provides a final validation of the design concept. Environmental tests will serve to fill in the information gaps left from component tests such as the transmissability levels associated with the total system assembly and mounting hardware.

7.6 SYSTEM TESTING OF THE ATTITUDE CONTROL SYSTEM

The assembly of all the subsystems of the ACS into a complete system will be required to prove overall system performance and compatibility during and after exposure to the applicable environments. These tests will be similar to those performed on the components and assembly level, however not as extensive since all of the components will have been tested individually.

The complete system assembly will provide a means of proving the dynamic performance of the ACS. This will be accomplished by using a three-axis air-bearing test facility. The complete system will be mounted and statically balanced. The system will be activated and exercised in a manner similar to that expected during a mission. That is, it will be commanded to perform reorientation maneuvers, limit cycle performance and simulated thrust vector control. The latter will be accomplished by providing disturbance torques to the air-bearing test bed and measuring the ability of the hot-gas TVC system to maintain proper orientation. The major difficulty of this type of testing is the maintenance of static balance for extended periods and over large angular excursions. However, if these limitations are recognized and either compensentated for (automatic balancing) or accepted as test limitations, valuable test data are available from this type of testing, and the dynamic response characteristics of the overall system can be verified. A precision air-bearing table is required for these tests.

8.0 PARACHUTE FLIGHT TESTS

8.1 TEST REQUIREMENTS AND OBJECTIVES

8.1.1 Rational for Flight Testing

Examination of the operational requirements for a Mars sub sonic descent parachute has led to the following conclusions:

1. The status of parachute technology is inadequate for the Mars application and a pre-Voyager technological development program will be necessary.

2. Ground test techniques are inadequate for both the pre-Voyager technological development and Voyager development programs, and must be augmented with flight tests in the Earth's atmosphere.

The technological status of the parachute is inadequate because the tenuous atmosphere of Mars requires parachute deployment at very low dynamic pressures (4 lb/ft^2), for which very limited experience exists and for which present analytical techniques are not applicable. This uncertainty is particularly important because many facets of the mission and system design, are significantly dependent on the parachute performance capability including allowable entry weight. Until the uncertainties are removed by flight system tests, formulation of authoritative mission and design concepts is not possible.

The ground test techniques are inadequate because of scaling limitations and infinite mass effects. Scaling effects associated with fabric porosity, flexibility, flow characteristics, etc., limit scaling to approximately one-tenth in parachute area and payload mass. This means that the scale model should be at least 27 feet in diameter, based on the reference design parachute diameter of 81 feet. No existing wind tunnel can accommodate this large a diameter at the correct flow ($M = 1.2$) and dynamic pressure (4 lb/ft^2). Sleds, whirl towers and the like cannot simultaneously simulate $M = 1.2$ and $q = 4\ lb/ft^2$ because the sea level atmosphere is too dense.

Infinite mass effects refer to the effects of fixed tiedown of the parachute shroud lines in ground testing (i.e., wind tunnel). Under real conditions the shroud lines are attached to a finite mass (the payload) and there is a mutual interaction between the dynamics of the payload and the dynamics of the parachute. As a consequence, fixed tiedown conditions can produce invalid results, particularly in canopy inflation, opening shock loads and parachute/payload stability.

The wake of the blunt vehicle could affect parachute performance and such wake effects must be evaluated. However, existing transonic wind tunnels limit the maximum vehicle model diameter to about one to two feet. Since this diameter would be scaled from the reference diameter of 15 feet, reliable wake measurements cannot be made.

8.1.2 Scale of Tests

The large size and weight of the entry vehicle and parachute make it attractive from the cost viewpoint, to conduct flight tests with subscale parachutes and payloads. In the pre-Voyager technological program the majority of the recommended tests are one-tenth scale. In addition, two full-scale tests are recommended to check scaling and blunt vehicle wake effects. One-quarter scale tests were also examined to determine the cost increment over one-tenth scale tests. The purpose of one-quarter scale tests would be to improve on limit scaling (one-tenth) and also to provide backup data if the one-tenth scale is attempted and found undesirable.

In the Voyager development program the parachute test program is equally divided between one-tenth scale and full-scale tests. There is a greater demand for full-scale tests because of the need for testing operational prototype hardware.

8.1.3 Deployment Conditions

The parachute deployment envelope is presented in Figure 3 on a plot of Earth altitude versus velocity. These parameters define both dynamic pressure and Mach number which can be plotted as contours of constant value. Both the Mars operational envelope and the recommended Earth test envelope are shown. The Mars envelope is bounded by a Mach number range of M = 0.7 to 1.2 and a dynamic pressure range of q = 5.0 lb/ft^2.

The Mars envelope results from entry and atmosphere uncertainties and limitations of the parachute initiation system. As discussed in paragraph 2.5.3, Book 6, Vol. V the firm constraints on deployment are a maximum Mach number of M = 1.2 and a maximum Mars deployment altitude of 27,500 feet. The latter constraint is imposed to limit the maximum descent time to satisfy relay communication requirements. These constraints are satisfied by initiating deployment at a time after peak deceleration which is a function of entry velocity and peak deceleration providing the limit altitude of 27,500 feet has been reached. The delay time t_d from peak to deployment is expressed as:

$$t_a = (0.041\,V_e - 225)\,g_{max}^{\,n},$$

where

$$n = 2 \times 10^{-5}\,V_e + 0.331.$$

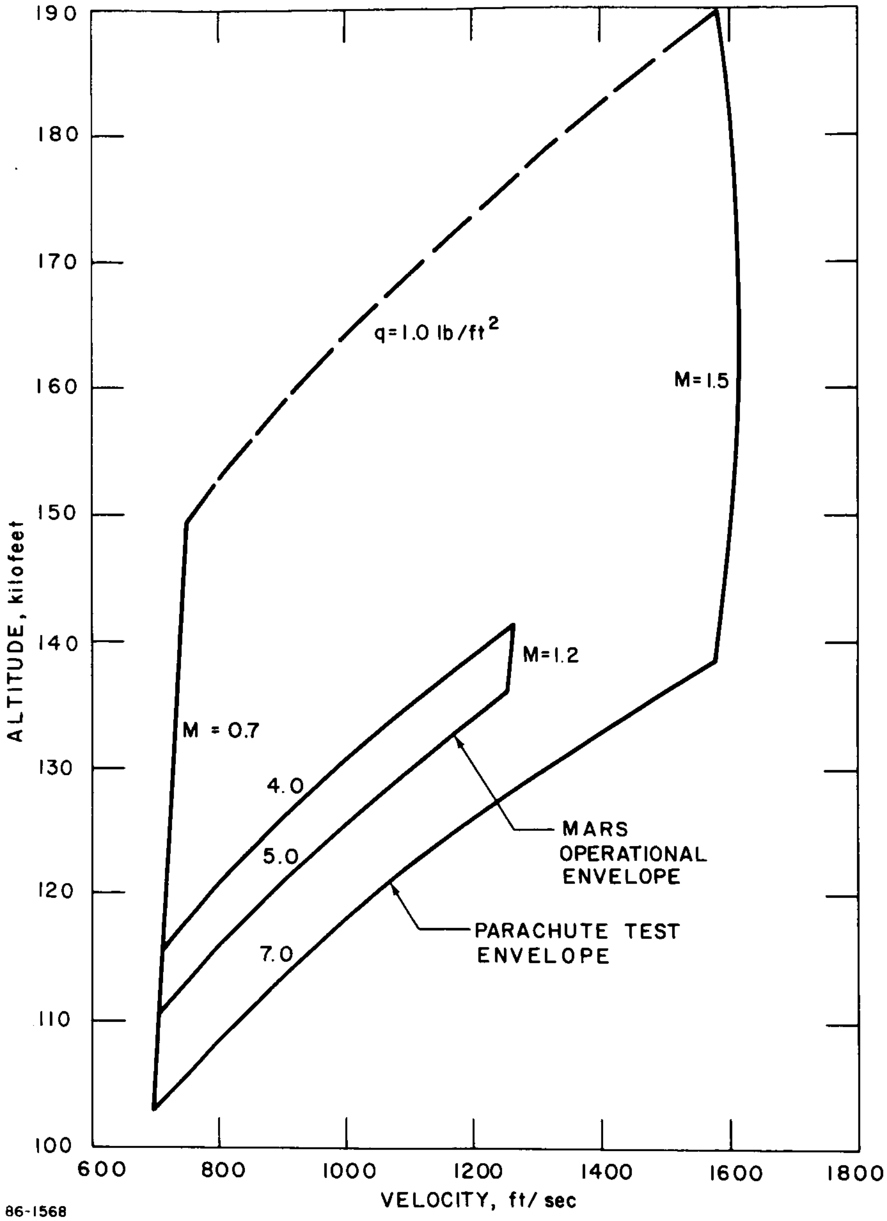

Figure 3 PARACHUTE DEPLOYMENT ENVELOPES -- EARTH FLIGHT TESTS AND MARS OPERATIONAL CONDITIONS

The Earth test envelope represents a somewhat arbitarary extension of the operational envelope. The purpose of the extension is to account for possible future variations in the Mars atmospheric model, the entry vehicle design concept and entry profile, and the type of deployment initiation system, all of which would affect the deployment envelope. Another purpose is to determine the minimum dynamic pressure and maximum Mach number limits of parachute performance. This determination is necessary in the establishment of design and performance margins and associated failure modes.

It should be noted that a problem in simulating the Mars atmospheric density may limit flight testing at the envelope corner which provides low Mach number and high dynamic pressure. This corner is at the lowest altitude in the envelope (approximately 100,000 feet). The Earth altitude at which the atmospheric density is equal to the Mars surface density is a little under 120,000 feet for the minimum density model, VM-7. Below 120,000 feet the density will be too high and descent velocities less than operational values, even though the deployment dynamic pressure is properly simulated. This problem can best be solved by testing at higher subsonic Mach numbers when high dynamic pressure conditions are desired. Variation of parachute performance over the subsonic regime should not be significant but can be checked by testing down to M = 0.7 at the lower dynamic pressures. In general, the transonic regime is of greater concern and most of the testing will be scheduled for this regime.

8.1.4 Payload Mass Simulation Versus Payload Weight Force Simulation

In the Mars operational flights, the entry vehicle shell is separated from the suspended payload immediately after the peak opening shock load of the parachute. The shell cannot be released earlier since its ballistic coefficient is smaller than the suspended payload coefficient and there would be danger of separation failure or post-separation collision. The shell is released as soon after deployment as possible in order to maximize deceleration of the suspended payload. In the full-scale parachute and full-scale parachute/separation flight tests, the boiler plate mockup of the entry shell, which has the correct mass, is jettisoned. For the subscale parachute flight tests the mass change due to the entry shell separation is simulated by jettisoning ballast. The suspended payload then has the correct mass, but since it is in the Earth's gravitational field its weight force will be larger than the Mars equivalent and descent velocities will be much higher than operational values. Additional ballast must then be jettisoned in order to reduce the suspended payload mass (by 61 percent) such that the weight force is correctly simulated. Jettisoning of this additional ballast is the subscale tests in not feasible since the remaining suspended weight is less than that required for the support systems such as telemetry, instrumentation, and power supply. Sufficient weight margin exists in the full-scale tests so that the additional ballast can be jettisoned and both the mass and weight force can be simulated on one flight. Weight force simulation in the subscale tests is accomplished by using a larger parachute on separate tests designed to operate at the proper area to weight ratio.

It should be noted that neither the mass simulation nor the weight force simulation is an ideal solution to the problem. For instance, for the weight simulation case, the descent velocity and flow field over the parachute is correct, but the parachute/payload mass combination is not simulated and stability results may be questionable. The problem is reversed in the mass simulation.

8.2 PRE-VOYAGER SUBSCALE PARACHUTE TESTS

8.2.1 Test Program

The selected approach for the pre-Voyager subscale parachute tests is a surface-launched Nike/Nike/Dart vehicle to provide the desired deployment conditions for a one-tenth scale parachute system. The Nike/Nike is a two-stage booster for the Dart test vehicle, an existing unpowered payload vehicle. The recommended test program consists of 32 flights at various deployment conditions within the test envelope described in paragraph 8.1.3. These flights would consist of both payload mass simulation and payload weight simulation.

A number of candidate test techniques were considered before the recommended approach was selected. The candidate techniques included various surface-launched and balloon-launched tests at both one-tenth and one-quarter scale. One-tenth scale, as previously mentioned, is regarded as the lower limit for reliable scaling. One-quarter scale testing was investigated to determine if it could be achieved without a significant increase in cost. The evaluation was reduced to three logical candidates: the selected program, a surface launched Honest John/Nike/Cree in quarter scale, and a balloon launched, newly designed test vehicle propelled in a climb by an Iroquois rocket. The superior candidate was not immediately apparent and a quantitative and qualitative comparison of their merits and deficiencies was required. The comparison was made on the basis of cost, test condition dispersion, launching ease, flexibility in adjustment of test condition and probability of test success. The results are summarized in Table XXIII. The comparison between the Nike/Nike/Dart and the Honest John/Nike/Cree was rather close, with the exception of cost, the Honest John/Nike/Cree being 1.5 times more expensive. The surface-launched vehicles are superior to the balloon-launched program in all factors except test condition dispersion and flexibility in adjustment of the test condition. The surface-launch dispersion disadvantage was not regarded as significant because the dispersions may be reduced by initiating parachute deployment based on real time deployment condition data. For instance deployment many be triggered by radio command in response to external tracking or

TABLE XXIII

COMPARISON OF SUBSCALE PARACHUTE TEST VEHICLES

	Nike/Nike/Dart	Honest John/Nike/Cree	Balloon Released Vehicle
Cost (Relative)	1	1.5	3.3
Dispersion: Δq (psf) at M_{NOM}	6	12	1
Dispersion: ΔM at q_{NOM}	0.2	0.3	0.1
Launching Ease	A	A	C
Flexibility in Adjustment of Test Condition	B	B	A
Probability of Test Success	A	A	B

by telemetered data on either dynamic pressure or Mach number. In a given flight deployment can be initiated at the desired dynamic pressure and the error in Mach number accepted. The role of the two parameters can be reversed in another flight. If accuracy in both test conditions is required in the same flight, the deployment can be triggered to equalize the dispersion in each condition. The balloon approach has greater flexibility in adjustment of test condition because the release altitude of the rocket propelled vehicle can be varied. This represents an additional adjustment parameter which is easily implemented without configuration changes. This advantage is not significant because the surface launched vehicles have adequate flexibility by varying launch angle, time delay between staging, ballasting, and aerodynamic drag. In the remaining factors of cost, launching ease, and probability of success, the surface launched programs are definitely superior to the balloon technique. The balloon program is more than three times as expensive as the Nike/Nike/Dart. The balloon and its control equipment are more expensive than the Nike/Nike booster, and the newly designed balloon test vehicle has greater development costs than the existing Dart vehicle. Launching ease of the surface launched vehicles was judged considerably superior to the balloon launch because the surface launched program requires less support equipment and personnel and is not subject to the same weather constraints as the balloon program. The balloon program requires mobile launch equipment with launch sites selected on the basis of prevailing winds. Launches must be delayed until the right combination of safe ground winds and desired winds aloft occur. As a consequence, more personnel and support equipment are required and launching delays are more frequent. The comparison of probability of test success was based on a qualitative consideration of test histories of both approaches. The flight record of the surface launched vehicle is very good. Balloons appear to have a greater number of failures due to wind damage during launch and quality control failures (leaks) after launch.

The recommended flight test program can be divided into three blocks of tests. The prime objective of the first two blocks will be configuration screening. The last block is for additional tests of the selected configuration. The first block (twelve tests) will consist of three different types of parachutes: ring-sail, extended scaled, and annular. All tests shall utilize weight force simulations and each configuration will be tested at four deployment conditions. The four conditions will be at the extremities of the Mars operational deployment envelope. It is anticipated that one of the candidates would be eliminated on the basis of the results of the first block of tests. The second block (eight tests) will consist of the two remaining candidates; each configuration tested at four deployment conditions and all tests using payload mass simulation. These tests will also be at the extremities of the Mars operational deployment envelope.

As a result of these tests, reference configuration can than be selected. The third block will consist of 12 tests of the selected configuration at various deployment conditions with both payload weight force and payload mass simulation. Two canopy geometries and two suspension geometries will be evaluated. Some of the deployment conditions will extend beyond the Mars operational envelope to higher Mach numbers and lower dynamic pressures in a search for critical performance limits.

8.2.2 Launch Vehicle Configuration

The launch vehicle configuration for the one-tenth scale parachute tests consists of two Nike solid rocket stages and a payload interface adapter as illustrated in Figure 4. Both stages are stabilized by aerodynamic fins which are canted to produce vehicle spin. No active system is used for flight path control. Second-stage and payload separation are accomplished by differential aerodynamic drag between the stages being separated. A clamping device locks the payload to the second stage until second-stage ignition to prevent inadvertent payload separation at first stage burnout. Second-stage burnout and separation occur at an altitude of approximately 30,000 feet after which the fin and spin stabilized test vehicle coasts to the deployment altitude.

Flight path adjustments to achieve various altitudes and Mach numbers within the deployment envelope are achieved by varying time delay between staging, launch angle, ballast in second-stage and drag devices in approximately that order of preference.

The Nike/Nike is capable of boosting a 200-pound, 9-inch diameter test vehicle to 170,000 feet at $M = 1.5$ which covers most of the high energy end of the arbitrarily selected deployment envelope. This capability exists for launches from the White Sands Missile Range or equivalent elevations. If launched from sea level the capability extends to approximately 150,000 feet at $M = 1.5$.

8.2.3 Test Vehicle Configuration

In order to achieve the desired payload/altitude performance the test vehicle must be a slender, low-drag configuration with the maximum diameter restricted to about 9 inches. Canted fins for aerodynamic stability and to maintain the launch vehicle induced spin rate will also be required. In the interest of cost, schedule and reliability factors, an existing vehicle such as Space Data Corporations' Dart vehicle is recommended. This is a versatile vehicle in which relatively standard fuselage sections such as a telemetering housing, beacon and programmer housing, payload housing, nose cone, etc., may be arranged in various tandem combinations as required. For the purposes of this test the configuration would consist of (in order from base to nose) aerodynamic fin section,

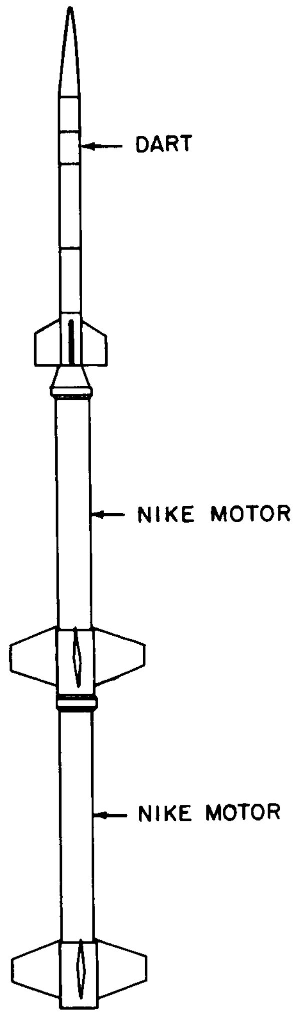

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Figure 4 NIKE/NIKE/DART 1/10- SCALE PARACHUTE FLIGHT TEST CONFIGURATION

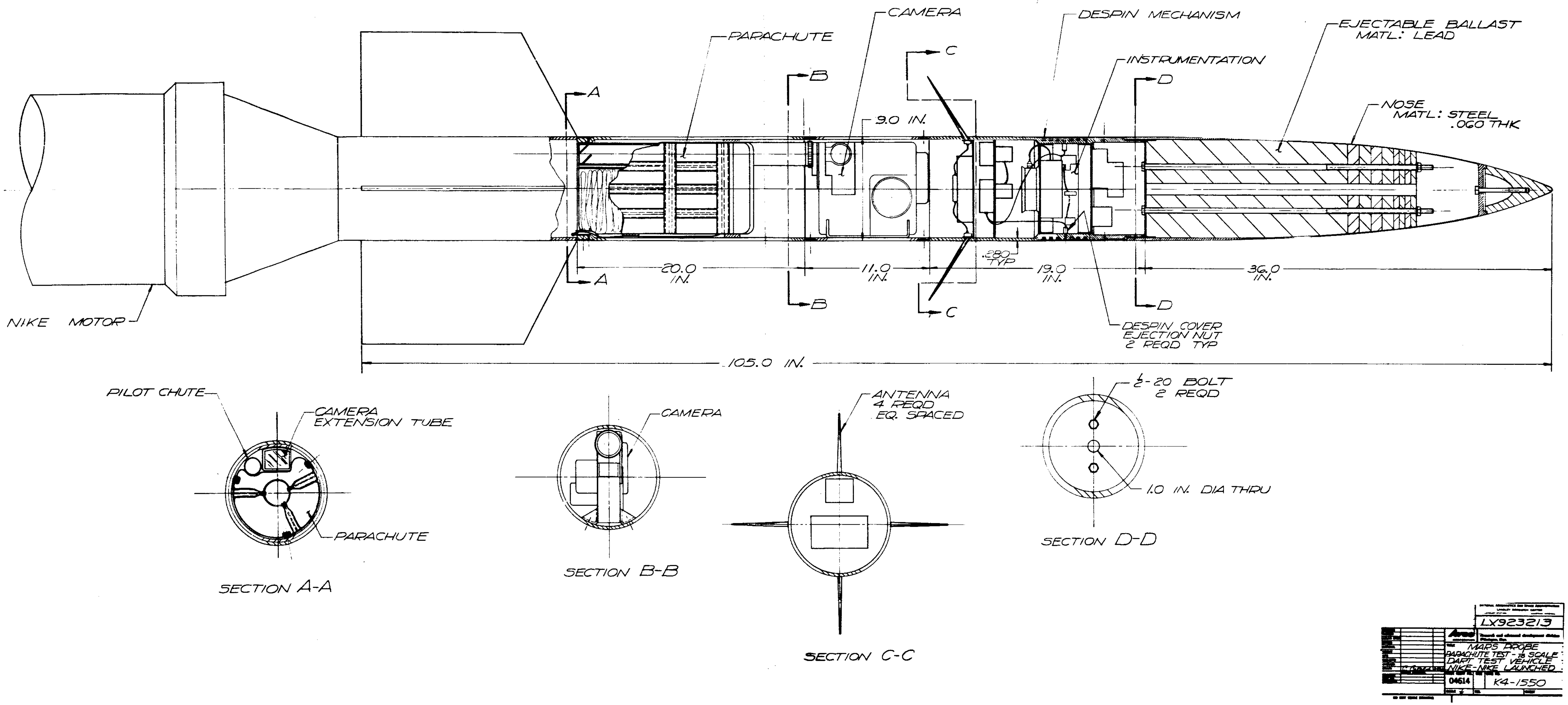

Figure 5 INBOARD PROFILE OF DART TEST VEHICLE -- 1/10-SCALE PARACHUTE FLIGHT TEST

test parachute housing, camera housing, telemetry housing, beacon and programmer housing, and a jettisonable nose cone containing ballast. The tandem order of these fuselage sections is based on functional requirements. The parachute is in a rearmost location to permit deployment from the base rather than the side of the vehicle. The fin section would be a logical housing for the parachute, but it is obstructed by internal interface structure which mates with the Nike adapter. The fin section is ejected prior to parachute deployment. The camera must be next to the parachute housing for unobstructed viewing of parachute deployment. The location requirement for the nose cone is obvious and the last two housings, telemetry and beacon, are installed in the remaining space, between the nose cone and camera housing. An inboard profile of the vehicle is shown in Figure 5. If the vehicle is flown over water, flotation gear and recovery aides such as a SARAH beacon and dye markers must be added to the configuration. This equipment would be packaged in a fuselage section inserted between the beacon housing and the nose cone. After nose-cone ejection the recovery housing base is exposed for easy deployment of the flotation and recovery aide equipment.

Ballast is installed in the nose cone and is jettisoned with the nose cone immediately after parachute deployment to simulate the mass change due to the entry vehicle shell during the operational flight. Total jettisoned weight is 101 pounds. The nose cone is jettisoned at its base using an existing ejection system which has a history of successful operation. The base of the nose cone overlaps the next fuselage section with two O-rings providing a gas seal in the overlap. An explosive charge located in a gap between the overlaps and forward of the O-rings generates gas pressure which ejects the nose cone forward in the manner of a cylinder sliding off a fixed piston.

The spin rate required to stabilize the vehicle at reasonable coning angles may be quite large, as much as 500 rpm. Deployment of the parachute at spin rates this high is undesirable even with swivel attachments. Spin rates less than 100 rpm are acceptable. A yo-yo despin mechanism located just aft of the nose cone base is used to reduce the spin rate to 100 rpm or less. The exact value must be determined by a detailed analysis of coning divergence during despin and the possible effects of large coning angles on the evaluation of parachute performance. Triggering the yo-yo mechanism just prior to parachute deployment is a possible solution if this proves to be a problem area.

The subsystems and components of the test vehicle are tabulated below:

Vehicle structure and fins

FM/FM telemetry

Battery power supply

Programmer and control circuitry

Radar beacon

Command reciever

Main Parachute

Pilot chute

Camera

Tensiometer

Accelerometers

Ram and static pressure sensors

YO-Yo despin mechanism

Umbilical connector

Jettisonable ballast

The nose-cone weight is 23 pounds, to which must be added ballast to increase the jettisoned weight to 101 pounds which is the scaled weight of the entry vehicle shell. The remaining weight of the vehicle should not exceed 102 pounds which is the scaled weight of the suspended payload including the parachute. Estimates indicate that this weight limit can be satisfied unless water recovery equipment is required. The weight is marginal for this case. It should be noted, however, that these weight limits are arbitrary and deviations are possible for a number of reasons:

1. The full-scale entry weight of 2040 pounds is for the current concept of a Voyager capsule. The ultimate value may be significantly different.

2. The one-tenth scale factor is not an inflexible selection. Its selection was economically motivated. Smaller scale is not desirable, but larger scale to accommodate the weight of necessary support equipment is feasible. The only limit on scale increase is the payload/altitude capability of the Nike/Nike. Preliminary estimates indicate that the test vehicle weight can be increased above 200 pounds without recourse to a larger launch vehicle. Employment of a larger vehicle such as an Honest John/Nike increases cost by a factor of 1.5, a

magnitude whose reasonableness can be judged only from within the confines of budgetary realities.

8.2.4 Flight Sequence

The flight sequence is pictorially illustrated in Figure 6. The Nike/Nike is launched at an angle of approximately 80 degrees, the exact angle depending on range safety requirements, impact area availability, and the desired test conditions at altitude. The vehicle is launched from a zero-length launcher at the desired angle. Acceleration by the first-stage Nike is very high, 20 g or more. Velocity will increase rapidly and the vehicle will begin spinning almost immediately. First-stage burnout occurs about 3.5 seconds after launch. Stage separation occurs immediately due to differential drag between the stages. The vehicle coasts for a short duration, the magnitude being a variable depending on the deployment conditions desired. The second-stage Nike burn time is also 3.5 seconds and payload separation is produced by differential drag. The fin stabilized and spinning test vehicle then coasts towards the deployment altitude. Before reaching the deployment altitude the aerodynamic fins and its fuselage section are jettisoned to provide access for parachute deployment. This fin jettison occurs when the dynamic pressure drops to low values and enough before parachute deployment to reduce collision hazards. The deployment sequence will be initiated when the vehicle reaches either the desired Mach number or dynamic pressure or a judicious combination of both parameters. The initiation signal will originate either from radio commands based on ground tracking and/or telemetry data or from vehicle instrumentation. The deployment sequence will be automatic; despin activation, parachute deployment, and nose-cone ejection in rapid succession. More detailed analysis may indicate that the order of the first two sequences should be reversed or occur simultaneously, the objective being to minimize coning divergence before parachute deployment. After reaching terminal velocity, the parachute and payload descend to the surface. The vehicle is recovered for postflight examination of the parachute and camera film. Telemetry, phototheodolite and radar tracking data are recorded throughout the flight.

8.2.5 Alternative Test Method Considered - Balloon Launched One-Quarter Scale Test Vehicles

8.2.5.1 Test Program

Although the balloon launched technique is not recommended, the results of its study are reported because of potential interest in this test technique.

One-quarter scale test vehicles were considered and three modes of deployment after release from the balloon were investigated:

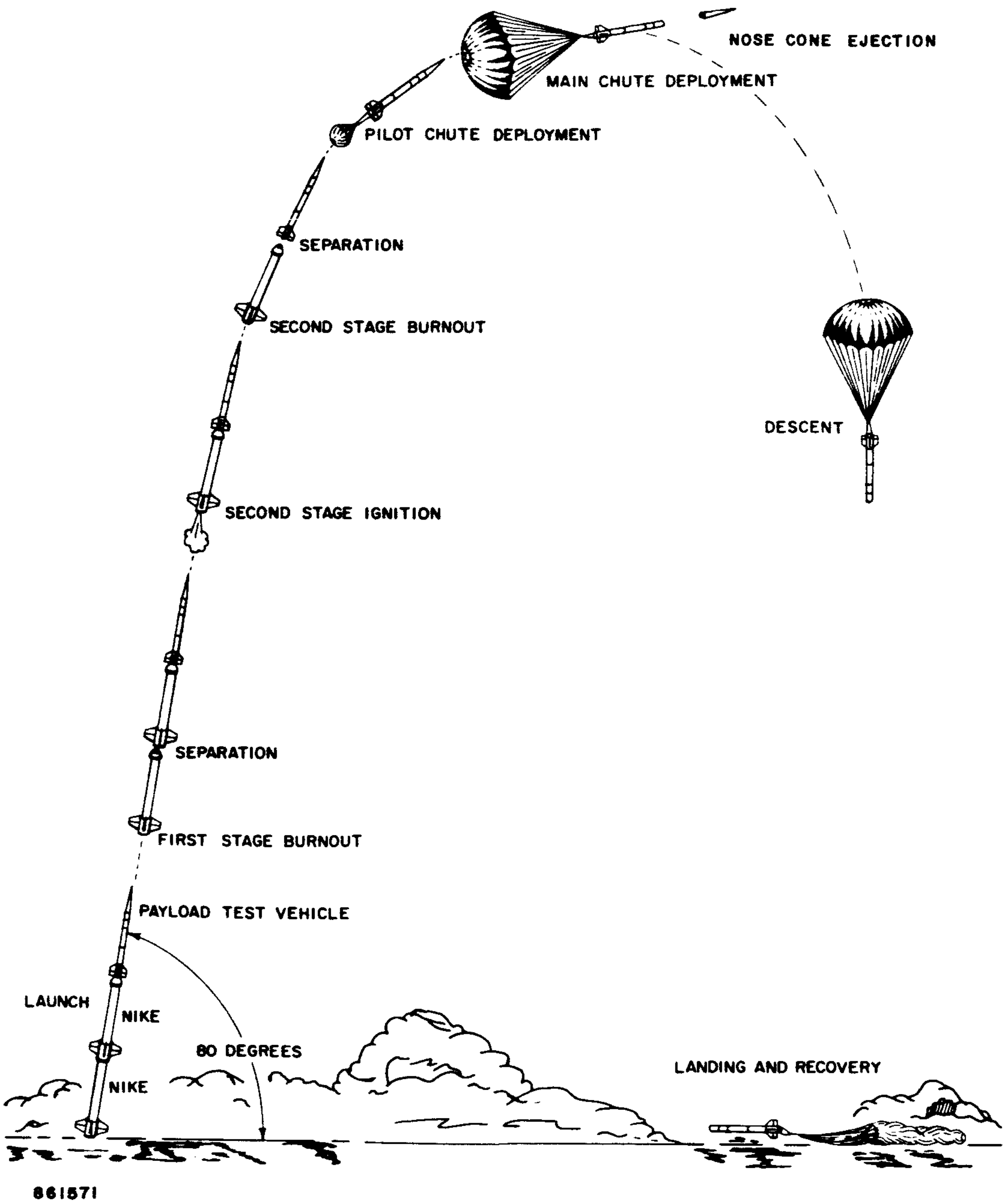

Figure 6 NIKE/NIKE/DART - 1/10-SCALE PARACHUTE TEST - PRE-VOYAGER AND VOYAGER-FLIGHT SEQUENCE

1. Rocket propelled climb

2. Unpowered free fall

3. Rocket propelled dive

The rocket-propelled climb was favored because it required lower release altitudes and hence smaller balloons. The unpowered free fall was of interest because of its simplicity. A speical test vehicle was designed which could accommodate all three modes of employment by using a cylindrical adapter to change the length of the fuselage. Two solid rocket motors of different length were used for the two powered modes, hence the need for adjusting fuselage length. An Iroquois rocket was used for the climb and a Sparrow sustainer motor for the rocket propelled dive mode. The test vehicle is a slender body which simulates only the scaled mass of the operational prototype and not its external shape. Flight path control is provided by vehicle spin. Both spin and despin are produced by solid rockets. The high-altitude balloon is a zero-pressure type, fabricated from Mylar film reinforced with bonded Dacron scrim. Balloon sizes for the climb mode will vary from 0.3 to 8 million cubic feet depending on the deployment condition.

The performance of the rocket climb vehicle is shown in Figure 7 which also demonstrates how various test conditions within the operational and test deployment envelopes are achieved. The vehicle trajectory is plotted in terms of altitude versus velocity. The rocket burn part of the trajectory is omitted for clarity. The ascent coast from burnout to apogee and the descent coast are shown for various launch altitudes. The spin stabilized attitude during rocket burn is 65 degrees (nose-up) for all the curves. Discrete time durations from burnout are marked on each trajectory. Time between burnout and deployment varies from 15 to 50 seconds depending on the desired test condition. Test conditions in the upper part of the deployment envelope can be achieved by launching at altitudes higher than those illustrated in Figure 7.

Unpropelled free-fall trajectories are shown in the same coordinate system in Figure 8. The vehicle reaches rather high velocities under gravitational acceleration alone. Release altitudes between 130,000 and 170,000 feet would be required to cover the entire operational deployment envelope. Required balloon volume, however, increases very rapidly at these altitudes. For instance, the required balloon volumes at altitudes of 130,000, 140,000, and 150,000 feet are, respectively 3.8, 8.5, and 28.5 million cubic feet. The largest balloon built to date was 13 million cubic feet; 28.5 million cubic feet may be achievable in the near future. Thus the

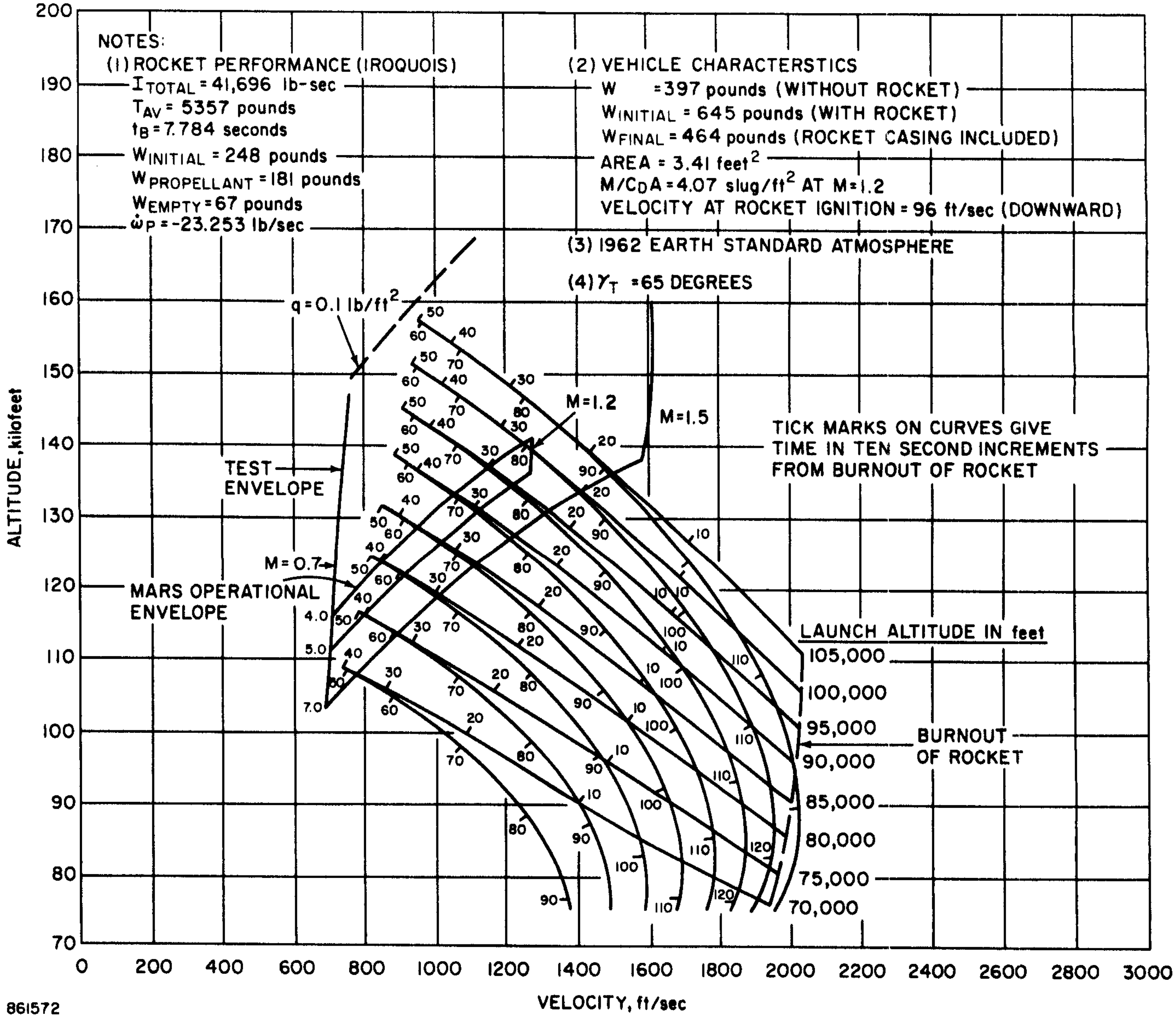

Figure 7 TEST VEHICLE TRAJECTORIES - 1/4-SCALE PARACHUTE TEST - ROCKET CLIMB FROM BALLOON RELEASE

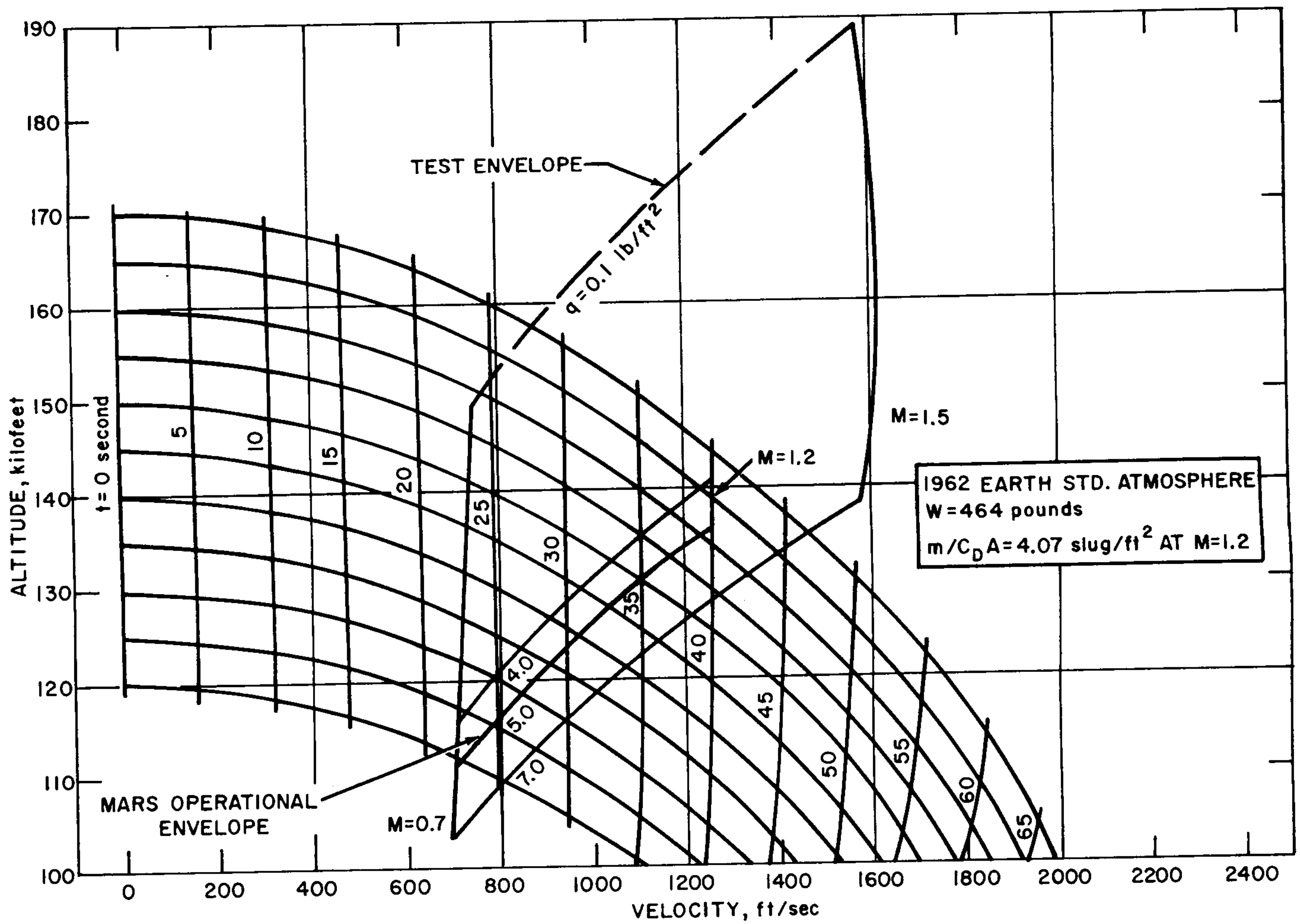

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Figure 8 TEST VEHICLE TRAJECTORIES - 1/4-SCALE PARACHUTE TEST - FREE FALL FROM BALLOON RELEASE free-fall mode has application only at the lower end of the deployment envelope.

The rocket propelled dive trajectories are also plotted in altitude versus velocity coordinates, Figure 9. Launches from 140,000, 145,000, and 150,000 feet altitudes are shown. Rocket burn times must be short (2.04 seconds) to complete burnout before the deployment condition is reached and some time must be provided to accomplish vehicle despin. This requirement will limit the minimum launch altitude to 140,000 feet if a test condition within the operational envelope is desired; burnout occurs about 2 seconds before the operational envelope for a launch from 140,000 feet. The upper limit on the use of the dive mode is about 150,000 feet as constrained by the required balloon size (28.5 million cubic feet). Thus the range of test conditions achievable with the dive mode is severely limited. In addition, the rocket climb mode provides the same test conditions with a much smaller balloon and hence the propelled dive mode is less desirable.

For the rocket climb mode horizontal range of the vehicle during powered and coast flight is presented in Figure 10 as a function of altitude. Ranges for trajectories launched from various altitudes are shown with the time from burnout indicated. Range from point of release to deployment will be between 3 and 8 nautical miles which would not cause insurmountable range safety problems, but would require control of the direction of launch.

8.2.5.2 Balloon Configuration

The balloon configuration consists of a zero-pressure balloon, recovery parachute, and balloon adapter. The balloon adapter is supported from the balloon by the parachute as shown in Figure 11. The parachute canopy is attached to the base of the balloon and the balloon adapter hangs from the parachute shroud lines. Balloon size required would vary depending on the deployment condition desired.

The balloon is helium filled and fabricated from bonded gores of Mylar film, reinforced with Dacron scrim bonded to the Mylar. A polyethylene balloon is not applicable because the total payload including balloon adapter approaches 1000 pounds which is too large for the strength characteristics of the polyethylene material. For the rocket climb after balloon release technique, release altitudes would vary between 70,000 and 130,000 feet requiring balloon sizes vary from 0.3 to 8 million cubic feet based on a balloon payload of 960 pounds which includes the test vehicle (660 pounds) and the balloon adapter (300 pounds). The adapter weight is its weight at release altitude; ballast, which is carried for control purposes, would have

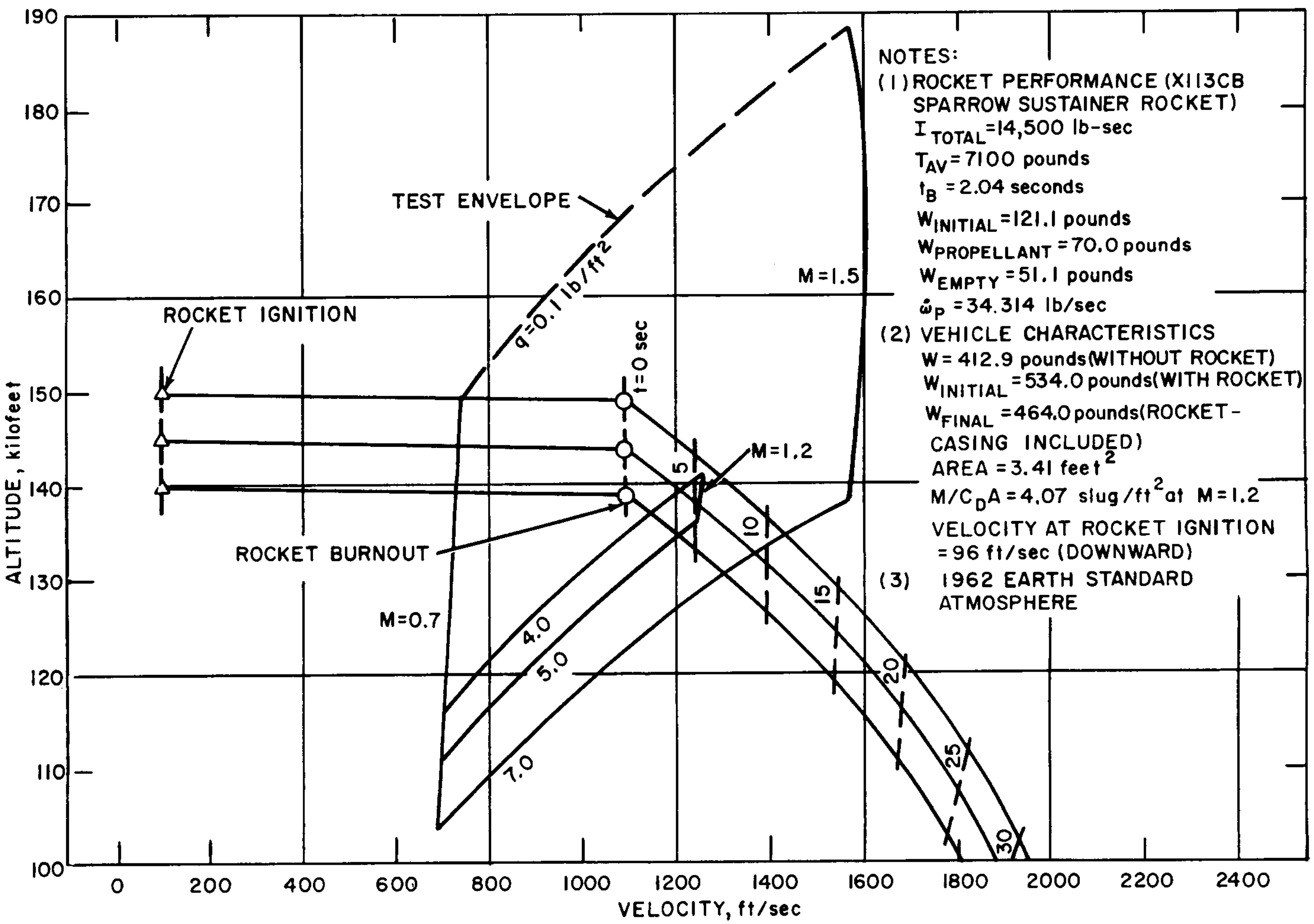

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Figure 9 TEST VEHICLE TRAJECTORIES - 1/4-SCALE PARACHUTE TEST - ROCKET DIVE FROM BALLOON RELEASE

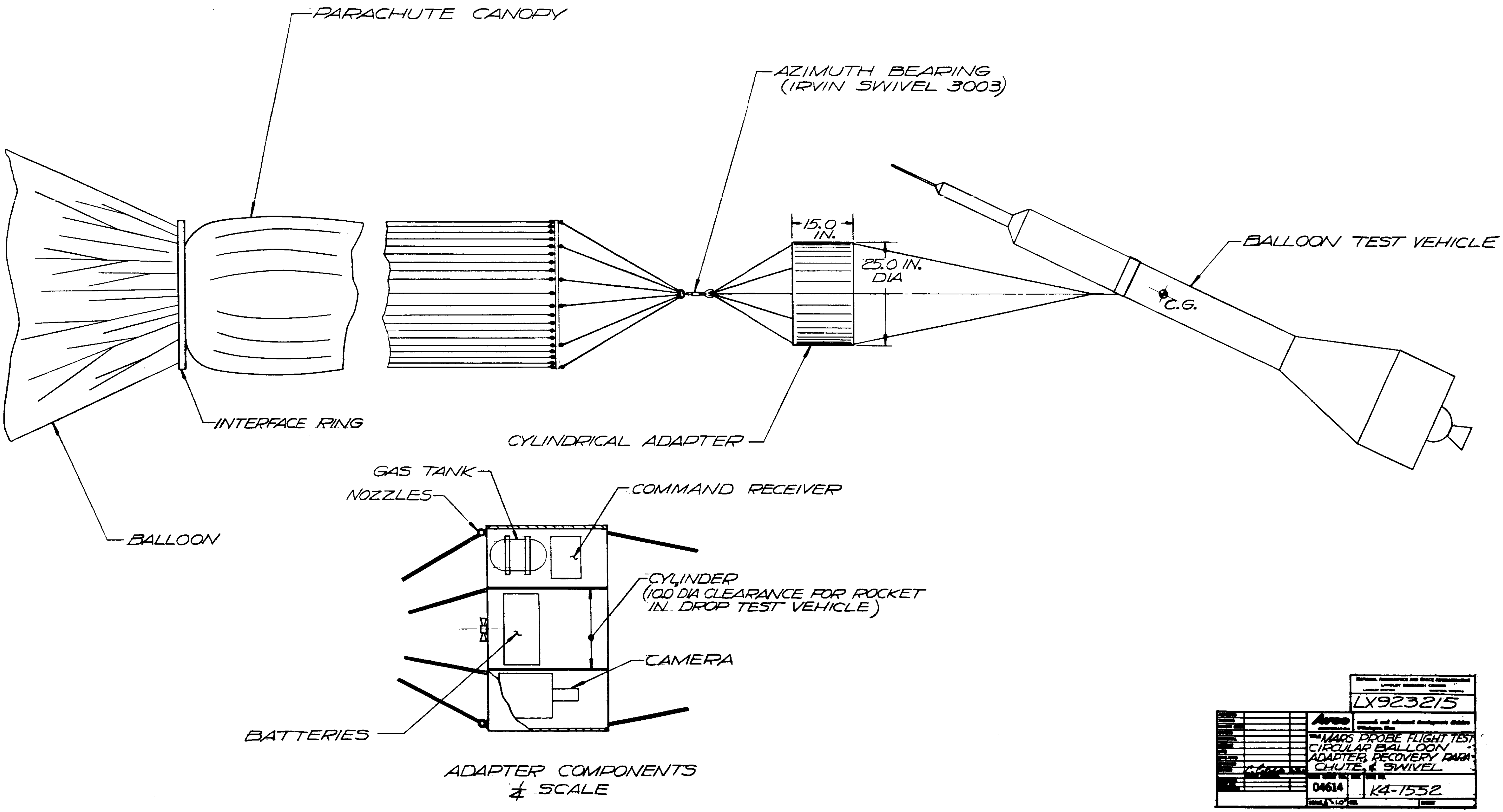

Figure 11 BALLOON ADAPTER CONFIGURATION -- 1/4-SCALE PARACHUTE FLIGHT TESTS

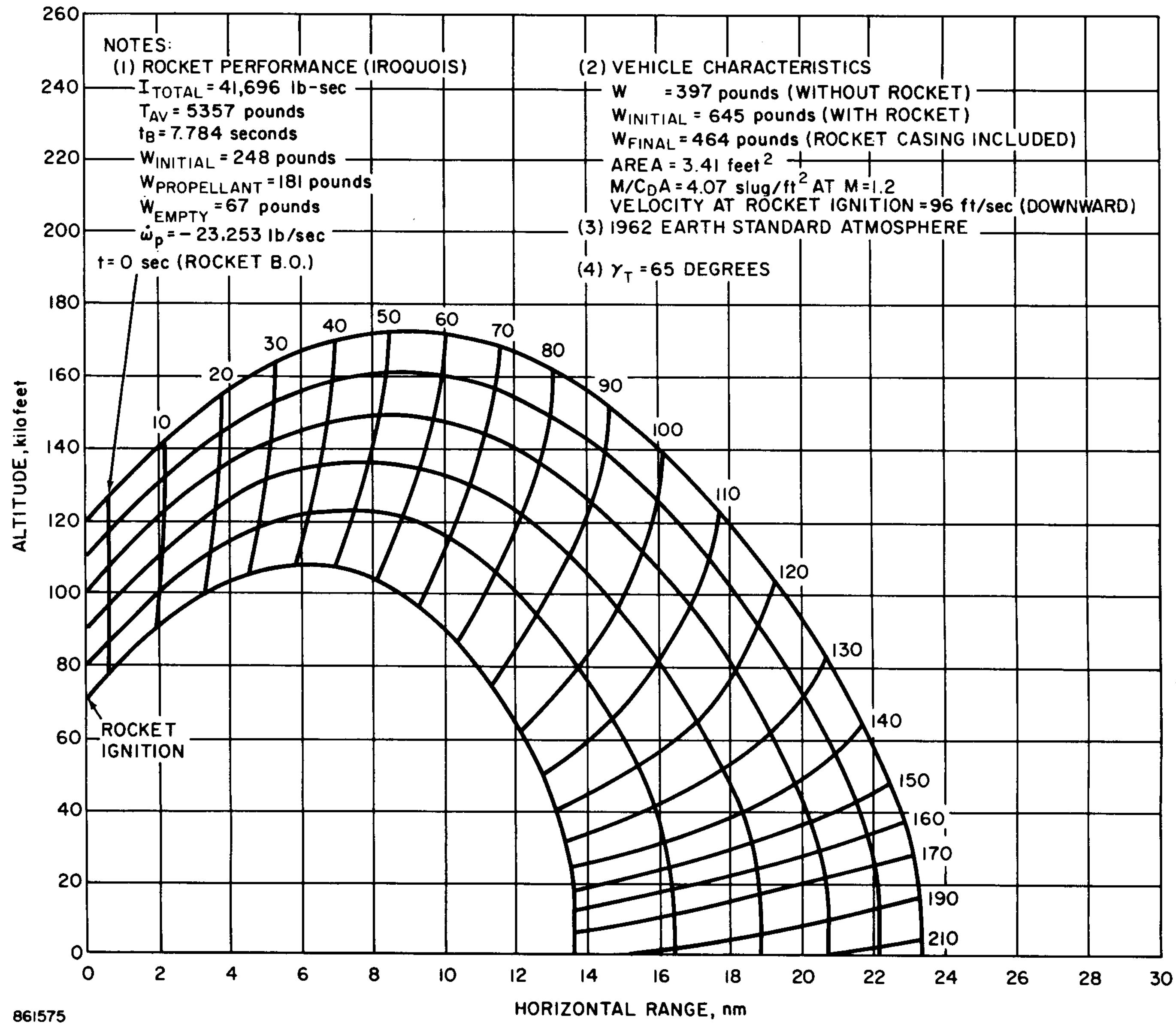

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Figure 10 TEST VEHICLE HORIZONTAL RANGE - 1/4-SCALE PARACHUTE TEST-
ROCKET CLIMB FROM BALLOON RELEASE

been previously jettisoned. The balloon volumes were also based on using Mylar scrim material of 0.35-mil thickness and weighing 3.5 lb/1000 ft^2 (Schjeldahl's GT-111 material). Balloon volume versus maximum altitude for various payloads is presented in Figure 12.

For the free-fall test technique120,000 feet to as high as 150,000 feet, if desired. Balloon sizes would be from 3.8 to 28.5 million cubic feet based on the same balloon material and a balloon payload of 760 pounds (300 pound adapter and 460 pound test vehicle). For the rocket dive from balloon release technique, release altitudes would be 140,000 to 150,000 feet, requiring balloon sizes from 14.5 to 28.5 million cubic feet. The test vehicle weight is 530 pounds and the total balloon payload is 830 pounds.

The balloon adapter is a cylindrical structure which supports and releases the test vehicle and houses the balloon control and support equipment. The adapter is hung from the recovery parachute shroud lines by an aximuth bearing. The bearing permits rotation of the adapter and test vehicle relative to the balloon and parachute to facilitate launching of the vehicle on a selected azimuth. Cold-gas reaction jets on the adapter provide the required torque and azimuth is sensed by a gyrocompass or equivalent instrument. The balloon equipment which is housed in the adapter consists of a command receiver and control circuitry, battery, camera altimeter transmitter, separation mechanism, ballast and jettison controls, gas valve controls, azimuth cold-gas reaction jet system, and umbilical connector. The command receiver and control circuitry receive and implement ground commands for telemetry calibrations, external/internal power switching, ballast ejection, balloon gas valving, azimuth control, camera initiation, test vehicle release and recovery parachute release. The battery supplies electric power to both the adapter equipment and the test vehicle prior to release. The camera is turned on just prior to vehicle release to record this event plus the subsequent spinup and rocket ignition. The ballast jettison controls and gas valve controls implement ground control of rate of ascent for compensation of unfavorable wind profiles and the effects of balloon adiabatic cooling. The umbilical connector between the adapter and the test vehicle contains power and signal leads, such as command, telemetry calibration, and diagnostic data. For the free-fall and rocket dive configuration the base of the test vehicle is supported at the base circumference by the cylindrical adapter which is the same diameter. Four ball-lock devices are used for attachment and release. For the rocket climb configuration the vehicle must be hung from the adapter at a steep nose-up attitude equal to the desired launch attitude. For convenience, the same cylindrical adapter is used but the vehicle is supported near its c.g., by four cables secured to the base of the adapter. The cables are attached to the vehicle by a two-strap rig around the

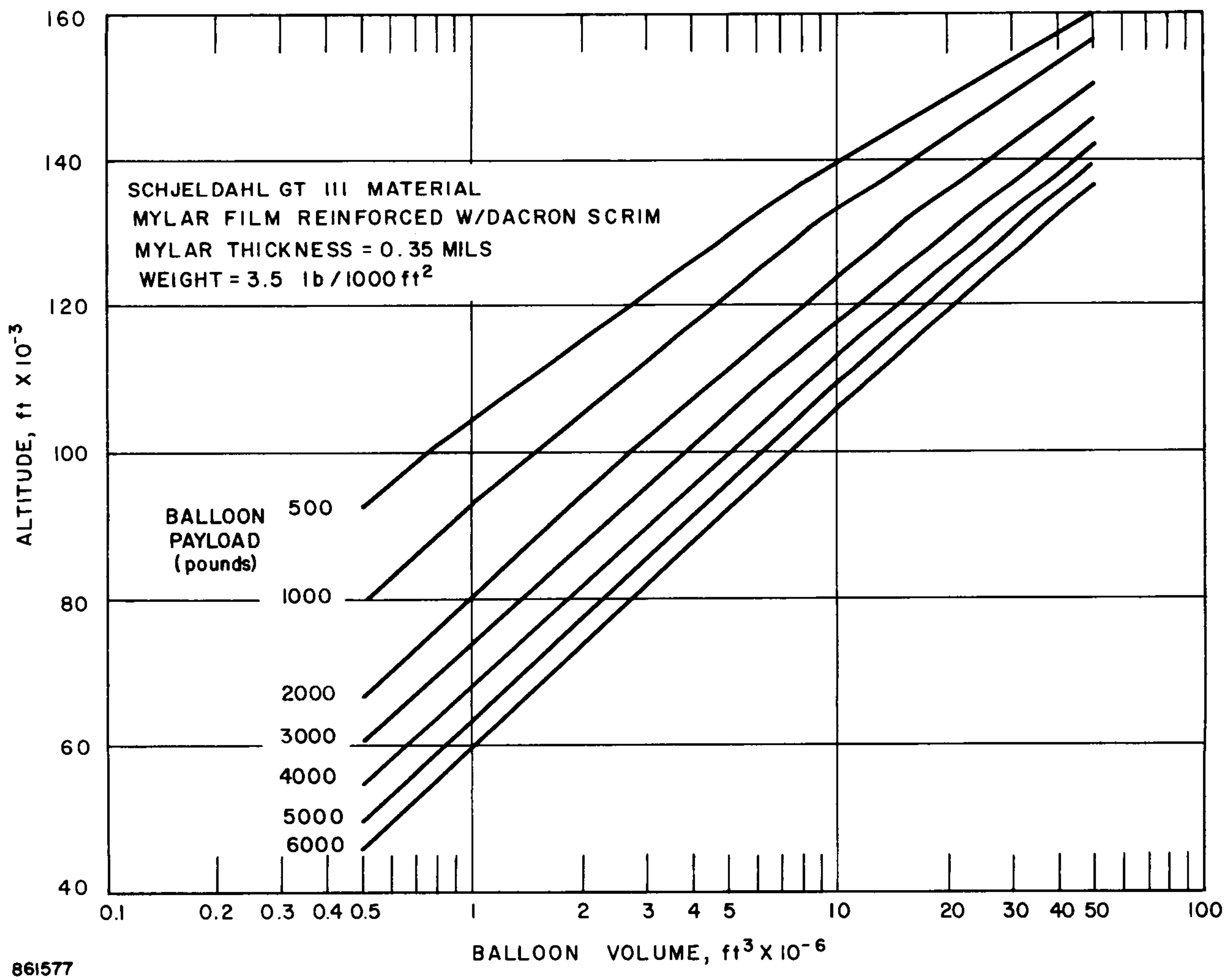

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Figure 12 BALLOON SIZE VERSUS ALTITUDE AND PAYLOAD CAPABILITY

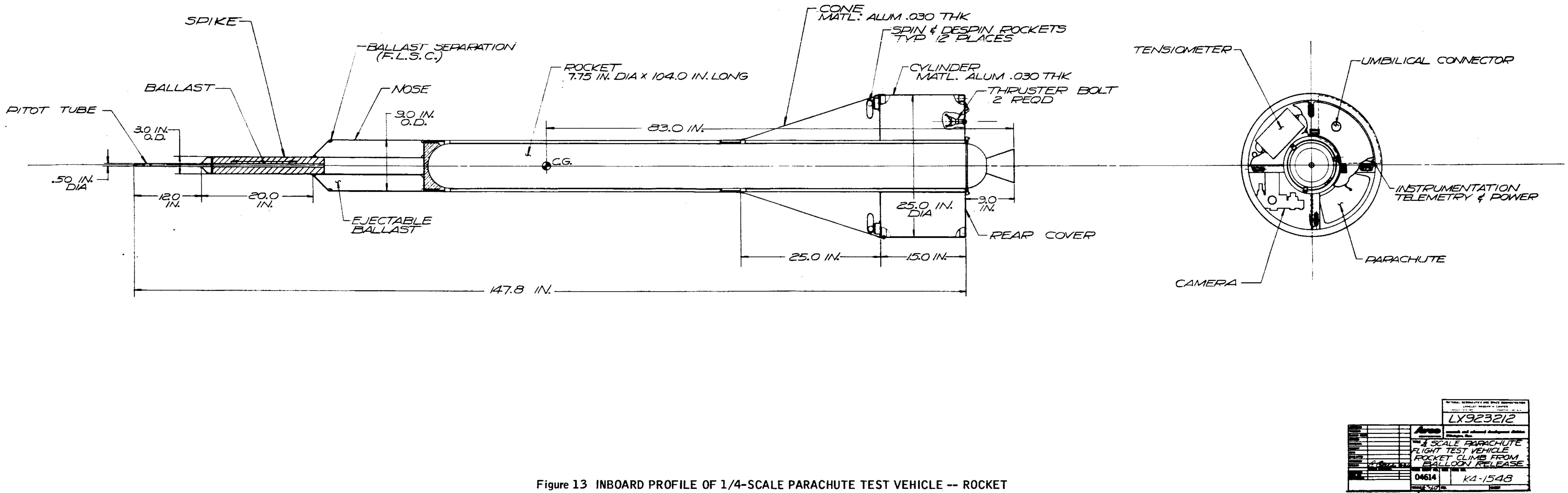

Figure 13 INBOARD PROFILE OF 1/4-SCALE PARACHUTE TEST VEHICLE -- ROCKET CLIMB FROM BALLOON RELEASE

fuselage. Release is accomplished by opening the fuselage straps with explosive bolts. The vehicle attitude is adjusted by sliding the straps along the fuselage until the centerline of suspension passes through the c.g., when the vehicle is at the desired attitude.

The canopy of the 65-foot recovery parachute is attached to the base of balloon by a metal ring two to three feet in diameter. The shroud lines are attached to the adapter through another ring of the same diameter. The purpose of this top and bottom ring support is to increase rigidity at these mechanical attachment so that the adapter and vehicle, when torqued in azimuth, will rotate relative to both the balloon and parachute. Without the ring support, adapter rotation will tend to wind up the shroud lines and/or canopy.

8.2.5.3 Test Vehicle Configuration

As previously discussed the rocket climb configuration was almost identical to the rocket dive vehicle. The former required a longer rocket which was accommodated by adding a cylindrical adapter to lengthen the fuselage of the latter. Inboard profiles of the climb and dive versions are shown in Figures 13 and 14, respectively. The free-fall test could use either configuration with either an expended rocket case or additional ballast.

The general arrangement of the vehicle was dictated primarily by two design criteria:

1. The expended rocket case cannot be jettisoned.

2. The parachute should be deployed from the base of the vehicle.

The expended rocket case could not be jettisoned because the time from burnout to parachute deployment was too short to accumulate safe dispersions between a jettisoned rocket case and the deployed parachute. This time interval is as short as 2 seconds for the rocket dive case and 15 seconds for the rocket climb. Deployment of the parachute from the base of the vehicle is an important design objective because it is a closer simulation of the operational deployment and a safer technique than deployment from a side hatch on the fuselage. The side hatch technique has a greater risk of the parachute or shroud lines fouling on the fuselage afterbody.

The two prime design criteria dictated selection of a rocket of small diameter and a fuselage diameter large enough to accommodate the rocket nozzle and parachute container in the base. Other equipment such as the tensiometer and camera must also be installed in the base. Minimizing the rocket diameter to maintain the fuselage

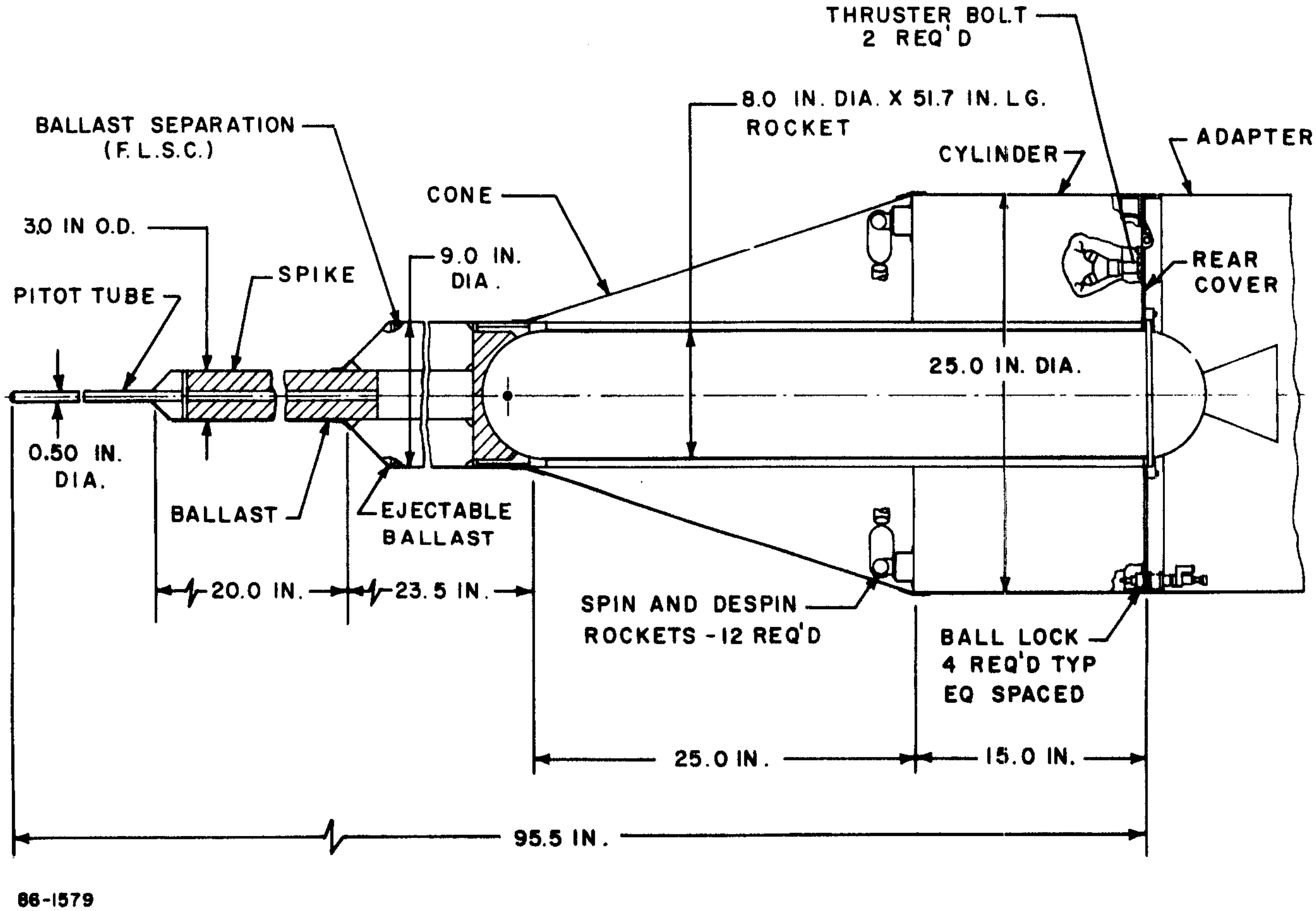

Figure 14 INBOARD PROFILE OF 1/4-SCALE PARACHUTE TEST VEHICLE-
ROCKET DIVE FROM BALLOON RELEASE

diameter at reasonable values resulted in the selection of a very long motor to achieve the required total impulse. Maintaining the base diameter of the fuselage throughout the length of the rocket would provide an internal volume far in excess of that required by the supporting subsystems. In fact a short (15-inch) cylindrical section at the base was adequate as shown in the inboard profiles. A conical section, largely empty, was added forward of the cylindrical section for streamlining. The remaining fuselage requirement was a cylinder in front of the rocket to house the jettisonable and fixed ballast and to provide structural support for the impact spike and the pitot-static tube. The jettisonable ballast, which simulates mass changes due to separation of the entry vehicle shell in the operational flight, consists of short or dust which is ejected by severing the nose of the fuselage with a linear-shaped charge as shown in Figure 13 and 14. The fixed ballast, which is required to increase the vehicle mass to the scale value, is installed inside the impact spike to aid forward location of the center of gravity. Thrust vector stabilization is provided by vehicle spinup with solid spin rockets. The high altitude at rocket ignition results in very low dynamic pressures throughout rocket burn and the use of canted aerodynamic fins for stability was judged infeasible. If the status of this alternative approach changes, this judgement should be examined quantitively. An approximate analysis of the spin dynamic revealed that required spin rates may be as high as 100 to 200 rpm. These rates are considered too high to accommodate by using a swivel between the parachute shroud lines and the support lines. Despin to a value lower than 100 rpm is desirable. Complete despin is not desirable because coning divergence may be too excessive. Coning half-angle and velocity vector pointing errors as a function of spin rate are presented in Figures 15 and 16 for various c. g., locations relative to the rocket nozzle. These results are approximate since aerodynamic effects were neglected. The error budget used for the analysis is shown in Table XXIV.

The spin and despin rockets are mounted internal to the conical section with the nozzles firing through large cutouts in the external skin. External mounting would also be suitable because of the low dynamic pressure environment.

An annular plate covers the base of vehicle except for cutouts for the rocket nozzle and the umbilical connector. This plate will give protection from the rocket plume and, when jettisoned just prior to parachute deployment, will provide simultaneously the necessary exposure of the pilot and main parachutes, tensiometer and camera.

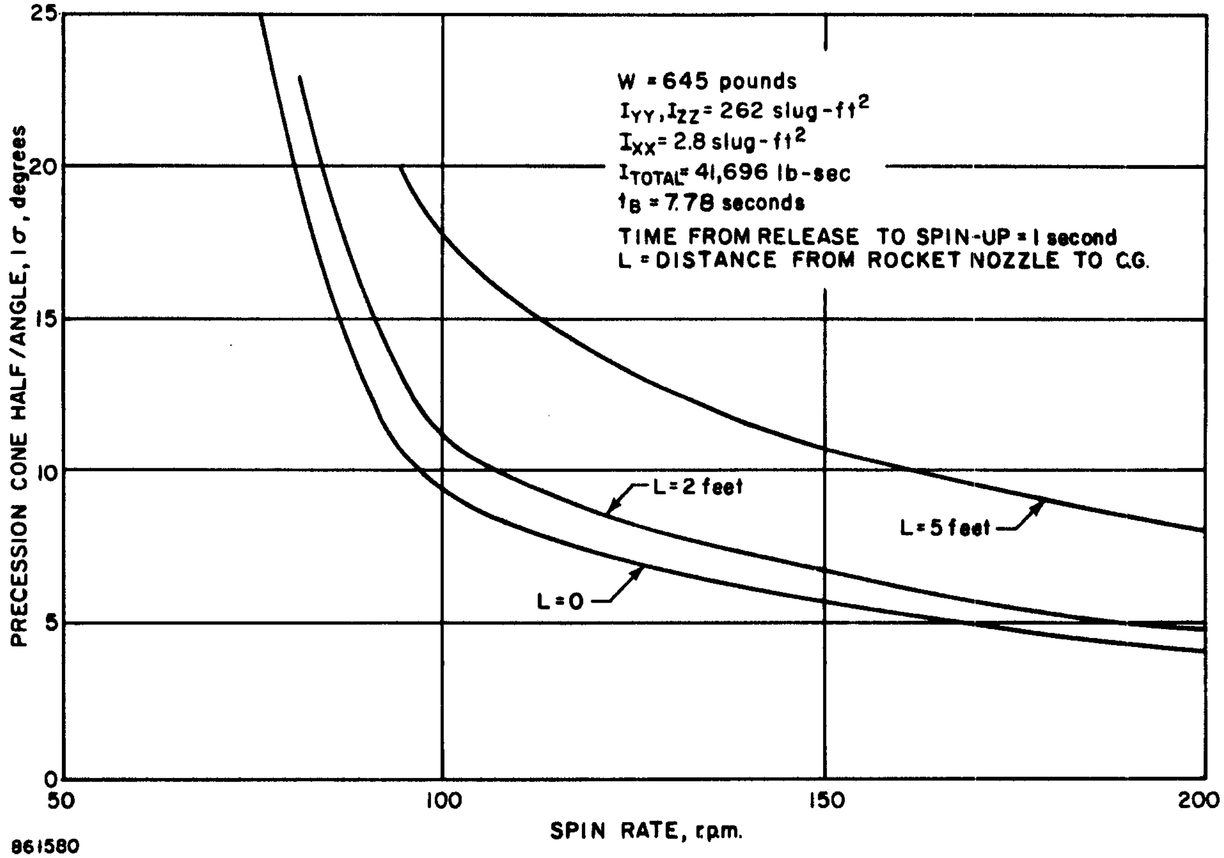

Figure 15 SPIN ANALYSIS - PRECESSION CONE HALF-ANGLE VERSUS SPIN RATE - 1/4-SCALE PARACHUTE TEST - ROCKET CLIMB FROM BALLOON RELEASE

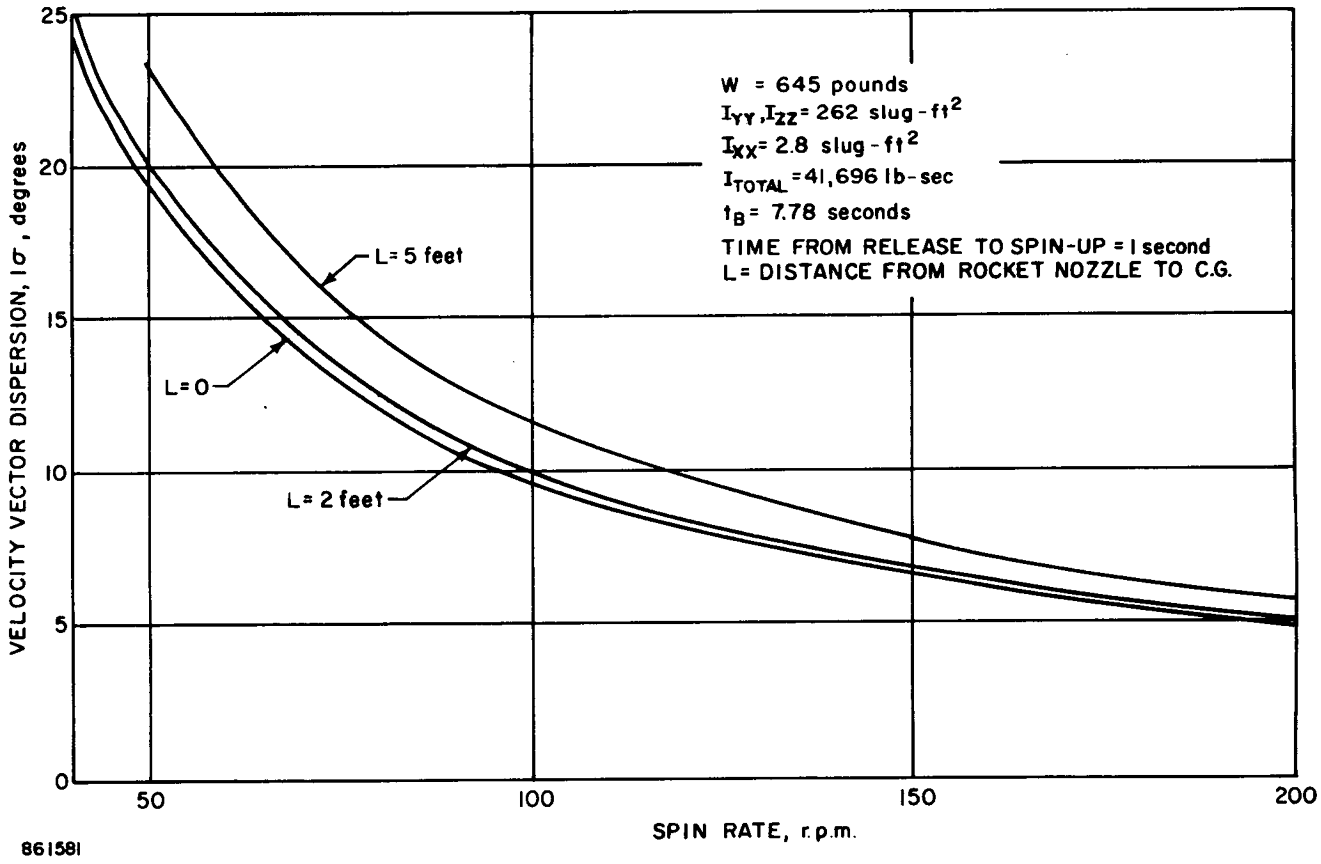

Figure 16 SPIN ANALYSIS - VELOCITY VECTOR DISPERSION VERSUS SPIN RATE - 1/4-SCALE PARACHUTE TEST - ROCKET CLIMB FROM BALLOON RELEASE

TABLE XXIV

SPIN ANALYSIS ERROR BUDGET - 1/4 SCALE PARACHUTE FLIGHT TESTS - ROCKET CLIMB FROM BALLOON RELEASE

Error Source	Value (1 sigma)
Initial attitude error at separation	1 degree
Spin rocket location error	0.042 inch
Spin rocket impulse error	1 percent
Angular misalignment of spin rocket thrust vector	0.167 degree
Thrust rocket location error	0.042 inch
Thurst rocket impulse error	1 percent
Angular misalignment of ΔV rocket thrust vector	0.167 degree
c.g., location error	0.0833 inch
Tipoff rates	0.5 degree per second

The important subsystems and components are:

FM/FM Telemetry

Battery Power Supply

Programmer and Control Circuitry

Main Parachute

Pilot Chute

Camera

Tensiometer

Accelerometers

Pitot-Static Tube

Iroquois Solid Rocket (or Sparrow Sustainer)

Spin and Despin Solid Rockets

Umbilical Connector

Impact Spike

Jettisonable and Fixed Ballast

The burnout weight of each vehicle is 464 pounds.* The initial weights of the climb vehicle (Iroquois rocket) and dive vehicle (Sparrow sustainer rocket) are 940 and 831 pounds, respectively.

*This weight is one fourth of a prototype vehicle weight of 1855 pounds, the reference design weight at the time this test vehicle was analyzed.

8.2.5.4 Flight Sequence

The prelaunch launch ascent and test vehicle release sequences are identical to the full-scale parachute test to be described in paragraph 8.3.4. The sequences subsequent to vehicle release are somewhat different and are discussed below. The flight sequences for the rocket climb mode and free fall mode are illustrated in Figures 17, and 18, respectively.

The vehicle is released from the ballon at various altitudes depending on the desired test condition within the deployment envelope. For the purpose of discussion, a typical case will be selected: release at 100,000 feet which will provide the deployment conditions, M= 1.2 and $q = 4\ lb/ft^2$. The test vehicle is released at an attiude angle of 65 degrees and spin stabilized immediately by spin rockets. After spin rocket burnout the Iroquois rocket is ignited-3 seconds after release. The vehicle starts climbing and accelerating and 7.8 seconds later, at burnout, is at an altitude of 106,000 feet and a velocity of 2030 ft/sec. Shortly after burnout the programmer ejects the vehicle aft cover which exposes the parachute canister, camera and tensiometer prior to deployment. The cover which protects the equipment from the rocket plume is jettisoned as early as possible after burnout to increase its dispersion and reduce the hazard of collision with the deployed parachute. The vehicle coasts in a climb and slowly decelerates. Twenty-six seconds after burnout it reaches an altitude of 140,000 feet and a velocity of 1250 ft/sec. The vehicle is now within the operational parachute deployment envelope (q = 4 psf, M= 1.2) and the deployment sequence is initiated by the programmer (or based on some other parametric, depending on the accuracy desired for the deployment conditions). First the vehicle spin rate is reduced from its initial high rate (200 rpm or greater) to a value less than 100 rpm by spin rockets. Immediately afterward, to alleviate coning angle divergence, the pilot chute is mortar ejected and pulls out the main parachute. Just after the peak opening shock load is sensed by an accelerometer, dust or shot ballast is jettisoned to simulate the mass change due to entry-vehicle shell separation in the operation case. The parachute and vehicle descend slowly to the surface and are recovered for post-flight evaluation. No additional ballast is jettisoned to simulate the Mars weight force (as would be the case for the full-scale vehicle) because the remaining payload weight allowance would be too small for the necessary equipment such as telemetry, instrumentation, etc. For weight force simulation, another flight is required using a larger parachute with the same payload weight. The balloon adapter is recovered by parachute in the same manner as the full-scale test.

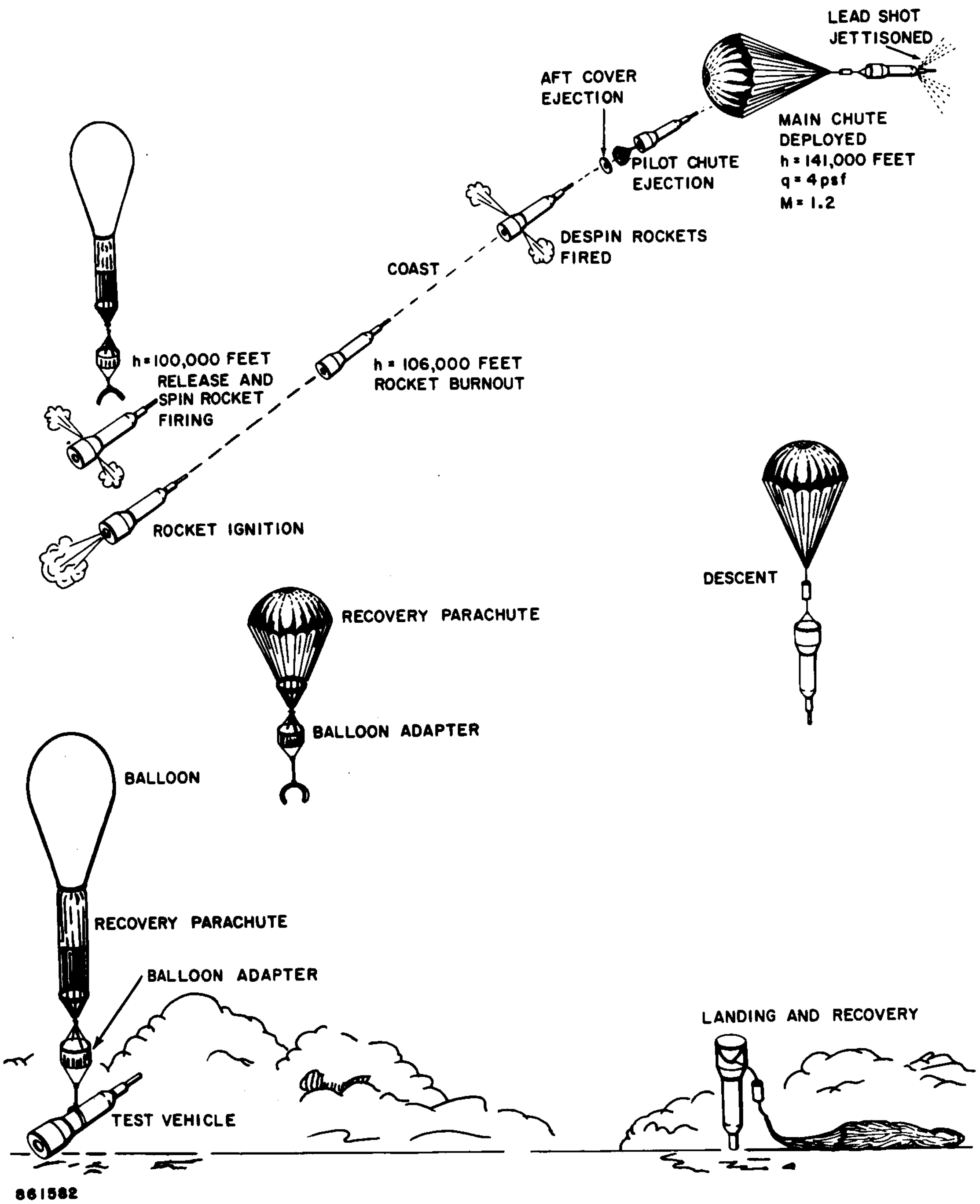

Figure 17 1/4-SCALE PARACHUTE TEST - ROCKET CLIMB FROM BALLOON LAUNCH - FLIGHT SEQUENCE

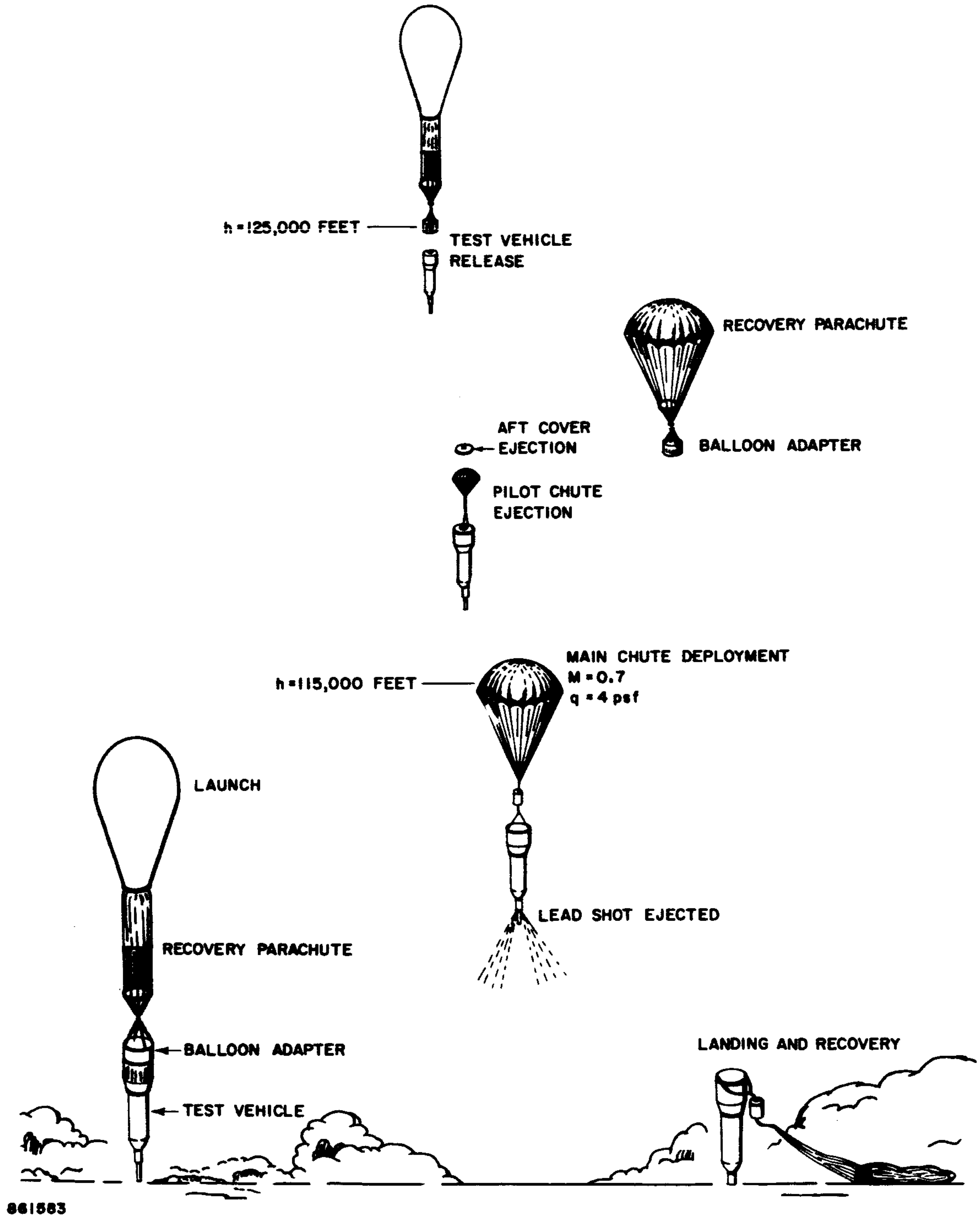

Figure 18 1/4-SCALE PARACHUTE TEST - FREE FALL FROM BALLOON - FLIGHT SEQUENCE

The flight sequence for free-fall mode is very similar to the rocket climb mode except that the vehicle is released nose down, it is not spin stabilized, and rocket propulsion is not used. In a typical case, illustrated in Figure 18, the vehicle is released at 125, 000 feet and free falls under gravitational acceleration. Twenty-three seconds later it reaches 117, 000 feet at approximately M = 0. 7 and q = 4 lb/ft^2 which is the low-energy end of the deployment spectrum.

The flight sequence for the rocket dive mode is even more similar to the rocket climb mode, the principal difference being that the vehicle is rocket-propelled straight downward instead of in a climb. In a typical case the vehicle is released at 145, 000 feet and, after spin stabilization, the Sparrow sustainer rocket is ignited. Burnout occurs 2 seconds later at 144, 000 feet and a velocity of 1100 ft/sec. The vehicle free falls and accelerates for 5 seconds to 138, 000 feet at which point the deployment condition, M= 1.2 and q = 4.5 lb/ft^2, is reached.

8.2.5.5 Deployment Condition Dispersion

A preliminary study of deployment condition dispersion of the ballon-launched vehicle was made for the comparative evaluation of the candidate test vehicles. The study also provided some insight into the relative dispersions in deployment conditions resulting from various methods of initiating deployment. Potential error sources for the dispersion were catalogued, the insignificant eliminated by inspection, and the significant analyzed in limited detail. An authoritative, in depth, analysis was beyond the scope of the study.

The error sources investigated were rocket total impulse (3 percent - 3 σ), spin-stabilized pitch attitude during rocket burn and coning half angle during rocket burn. The last two were obtained from the spin analysis report in paragraph 8.2.5.3 which included nine error sources (Table XXIV) hence the dispersion analysis is really based on more than three errors. The effect of the coning angle is to reduce the effectiveness of the rocket total impulse in accelerating the vehicle this reduction being an amount determined by the cosine law. The trajectory dispersion due to the total impulse error was computed to determine the deployment condition dispersion. The combined rocket and coning angle total impulse error is about 5 to 10 percent depending on the exact spin rate that is used. The effect on deployment is presented in Table XXV. The dispersions due to 5, 10, and 15 percent total impulse errors are presented. The results are expressed in terms of dispersions of six trajectory parameters when one of the parameters is at the correct value. For instance, when the dispersed trajectory is at the reference Mach number, the value of γ, q, t, h, and V are recorded as the dispersions for that reference parameter (Mach

TABLE XXV

TRAJECTORY DISPERSION RESULTS
-1/4 SCALE PARACHUTE TEST-ROCKET CLIMB FROM BALLOON RELEASE

Case No.	Constant Parameter	Δt (seconds)	$\Delta\gamma$ (degrees)	Δh (feet)	ΔV (fps)	ΔM	Δq (psf)
1.0 (-5 percent on t_B)	time	0	-0.997	-1656	+6.575	+0.009	+0.34
	altitude	-1.943	+1.146	0	-33.329	-0.030	-0.21
	velocity	-0.320	-0.655	-1385	0	+0.003	+0.24
	Mach No.	-0.460	-0.505	-1266	-3.029	0	0.20
	dynamic pressure	-1.167	+0.274	- 684	-17.571	-0.014	0
2.0 (+5 percent γ_T)	time	0	-1.716	+4367	-111.787	-0.110	-1.26
	altitude	+5.083	-7.088	0	+ 2.580	+0.004	+0.01
	velocity	+4.973	-6.984	+ 105	0	0	-0.02
	Mach No.	+4.956	-6.968	+ 120	- 0.388	0	-0.03
	dynamic pressure	+5.045	-7.049	+ 39	+ 1.610	+0.003	0
3.0 (-5 percent γ_T)	time	0	+1.053	-4911	-111.074	+0.120	+1.89
	altitude	-6.208	+7.984	0	- 3.602	-0.003	-0.03
	velocity	-5.999	+7.729	- 144	0	0	+0.02
	Mach No.	-6.010	+7.743	- 136	- 0.199	0	+0.02
	dynamic pressure	-6.070	+7.815	- 94	- 1.244	-0.001	0
4.0 (-10 percent on t_B)	time	0	-5.548	-9495	- 41.035	+0.056	+2.47
	altitude	-12.696	+9.858	0	-211.007	-0.198	-1.23
	velocity	- 1.869	-3.699	-7782	0	+0.019	+1.61
	Mach No.	- 2.820	-2.708	-6953	- 20.486	0	+1.24
	dynamic pressure	- 6.634	+1.939	-3907	-100.486	-0.086	0
5.0 (10 percent γ_T)	time	0	-4.280	+8169	-221.775	-0.220	-2.10
	altitude	+9.282	-13.659	0	+ 4.124	+0.006	+0.03
	velocity	+9.120	-13.527	+ 166	0	+0.003	-0.03
	Mach No.	+8.990	-13.421	+ 300	- 3.323	0	-0.07
	dynamic pressure	+9.204	-13.595	+ 80	+ 2.140	+ .004	0
6.0 (-10 percent γ_T)	time	0	+ 1.589	-10,327	+218.682	+0.230	+4.73
	altitude	-14.124	+17.775	0	- 7.448	-0.002	-0.05
	velocity	-13.538	+16.992	- 308	0	+0.004	+0.05
	Mach No.	-13.910	+17.489	- 112	- 4.730	0	-0.01
	dynamic pressure	-13.832	+17.385	- 153	- 3.741	+0.001	0
7.0 (-15 percent on t_B)	time	0	-11.996	-21,621	+103.230	+0.236	+8.38
	altitude	--	--	--	--	--	--
	velocity	- 4.498	- 8.112	-17,040	- 0	+0.035	+4.52
	Mach No.	- 5.925	- 6.733	-15,717	- 32.225	0	+3.61
	dynamic pressure	-14.748	+ 3.734	8,917	- 27.892	-0.190	0
8.0 (+15 percent γ_T)	time	0	- 7.950	+11,387	-326.954	-0.320	-2.67
	altitude	+12.713	-19.915	0	+ 4.943	+0.004	+0.03
	velocity	+12.531	-19.789	+ 197	0	+0.001	-0.04
	Mach No.	+12.490	-19.760	+ 242	- 1.129	0	-0.05
	dynamic pressure	+12.633	-19.859	+ 87	+ 2.759	+0.003	0
9.0 (-15 percent γ_T)	time	0	+ 1.704	-16,213	+320.329	+0.350	+9.05
	altitude	-27.256	+34.563	0	- 7.580	-0.003	-0.06
	velocity	-25.463	+31.940	- 355	0	+0.004	+0.06
	Mach No.	-25.910	+32.590	- 256	- 2.141	0	-0.02
	dynamic pressure	-26.277	+33.126	- 180	- 3.769	-0.002	0

number). The dispersions for the other reference parameters are also recorded. This information is useful, even though M and q are the prime factors of importance in deployment simulation, because the other parameters are candidates for deployment initiation. The data will show which are the best initiation parameters.

The deployment condition dispersion due to the third error, spin-stabilized pitch attitude during rocket burn (γT) (approximately 10 degrees) was also determined from trajectory dispersion and the results presented in the same manner in Table XXV. The 10-degree error (15 percent) in attitude produces a negligible dispersion in the deployment conditions (M and q) for all the initiation parameters except time. Therefore, unless time is used as the initiation parameter, the total deployment condition dispersion will be that due to the rocket impulse and coning angle errors.

8.3 PRE-VOYAGER FULL-SCALE PARACHUTE TESTS

8.3.1 Test Program

The selected program for the pre-Voyager full-scale parachute tests is a ballon-launched test vehicle which is rocket-propelled in a climb to the desired deployment conditions. The high-altitude balloon is a zero pressure type fabricated of Mylar film reinforced with Dacron scrim. The 6.5 million cubic foot balloon releases the test vehicle at 110,000 feet at a pitch attitude of 60 degrees. The vehicle is spin-stabilized at this attitude, and an Alcor rocket motor propels it to the deployment altitude, 140,000 feet. The purpose of the vehicle climb is to minimize altitude requirements (and hence volume) for the balloon. Rocket propulsion is required in any event to accelerate to the deployment Mach number (M= 1.2). The full-scale test vehicle is a boilerplate mockup with the external shape and mass characteristics of the prototype vehicle simulated. The recommended program consists of two tests at the high-energy end of the operational envelope (M= 1.2 and q = 4 lb/ft^2). Payload mass and payload weight will be simulated on both flights by jettisoning extra ballast during the parachute descent. The two flights should be scheduled early in the pre-Voyager program because their purpose is to verify the scaling validity of the subscale tests and to check possible blunt-body wake effects on the parachute performance.

The number of potential launch vehicles available for the full-scale test was severely restricted due to the large dimensions of the test vehicle (15-foot diameter). Only two launch vehicles were feasible: the Little Joe II and the balloon just described. Even the Little Joe II requires a hammerhead shroud because the test vehicle diameter is larger than the booster's. The balloon was a clear choice based on the factor of cost.

The performance of the test vehicle is shown in Figure 19 in which vehicle trajectories are plotted in terms of altitude versus velocity for launchings from various altitudes. All launch angles are 60 degrees. The powered part of the trajectory from balloon release to burnout is omitted for clarity. Ascent coast from burnout to apogee and descent coast from apogee are shown. As indicated, a launch altitude of 110, 000 feet will provide the desired deployment conditions during ascent for the two tests (M=1.2 and 2 = 4 lb/ft^2). Since a 60-degree launch angle is about the maximum safe angle to avoid collision with the ballon overhead, 110, 000 feet is approximately the minimum lauch altitude from which the desired conditions can be achieved. Horizontal range of the vehicle during the powered flight and coast periods is plotted in Figure 20 as a function of altitude for several trajectories launched from different altitudes. Time from burnout is indicated on the chart. For the reference trajectory, deployment occurs about 10 seconds after burnout or 5 nautical miles from the point of balloon release. This magnitude won't create insurmountable range safety problems but it is large enough to require control of the direction of launch.

8.3.2 Balloon Configuration

The balloon configuration consists of a reinforced Mylar, zero-pressure ballon, recovery parachute, and balloon adapter. The adapter is supported from the balloon by the parachute in the same arrangement described for the one-quarter scale balloon launched parachute tests. The configuration is illustrated in Figure 21.

The same type of balloon is recommended as that used for one-quarter scale tests and reference is made to paragraph 8.2.5.2 for a complete description of the balloon. The two full-scale, pre-Voyager flight tests will be launched from the same altitude, 110, 000 feet. The balloon size required for this altitude is 6.5 million cubic feet. The balloon payload is 3000 pounds which includes a 2700-pound vehicle and a 300-pound adapter.

The configuration of the balloon adapter structure is different to accommodate the larger dimensions of the test vehicle. The structure is a six-foot long triangular truss which provides a wider spread for the vehicle support cables. The two support cables are attached to a sway brace structure. The sway brace is a T-bar configuration (figure 21) which attaches to the vehicle with an explosive nut at only one point near the center of the long bar. Short pegs on the ends of the long bar and the T-bar key with hard point, tooling holes on the vehicle. The suspended attitude of the vehicle is adjusted by altering the relative length of the two support cables. The point at which the cables attach to the sway brace are altered at same to ensure that the suspension centerline goes through the vehicle c. g. The same balloon control and support equipment are used and these are mounted on the triangular truss. This truss is also supported from the recovery parachute

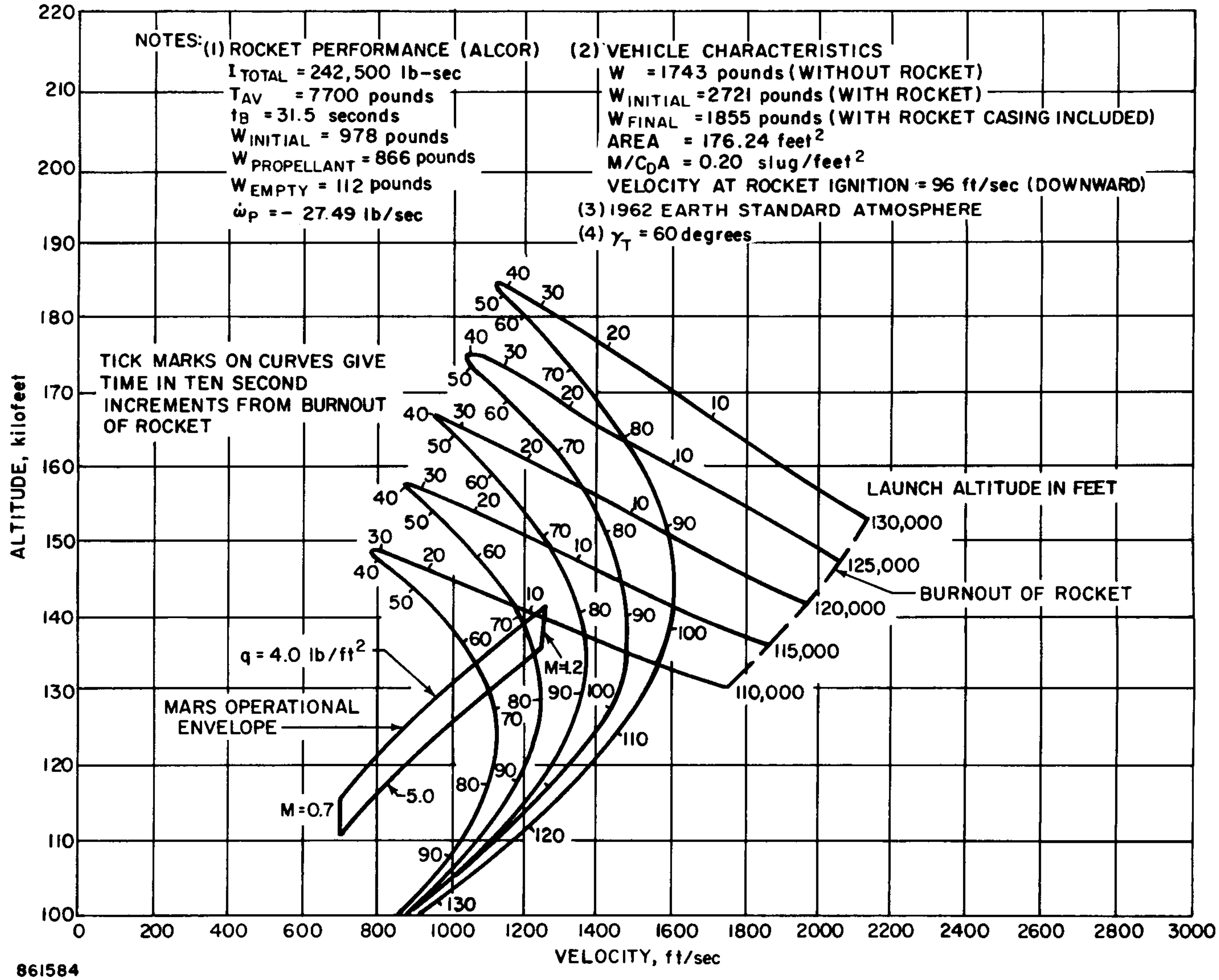

Figure 19 TEST VEHICLE TRAJECTORIES - FULL-SCALE PRE-VOYAGER PARACHUTE TEST - ROCKET CLIMB FROM BALLOON RELEASE

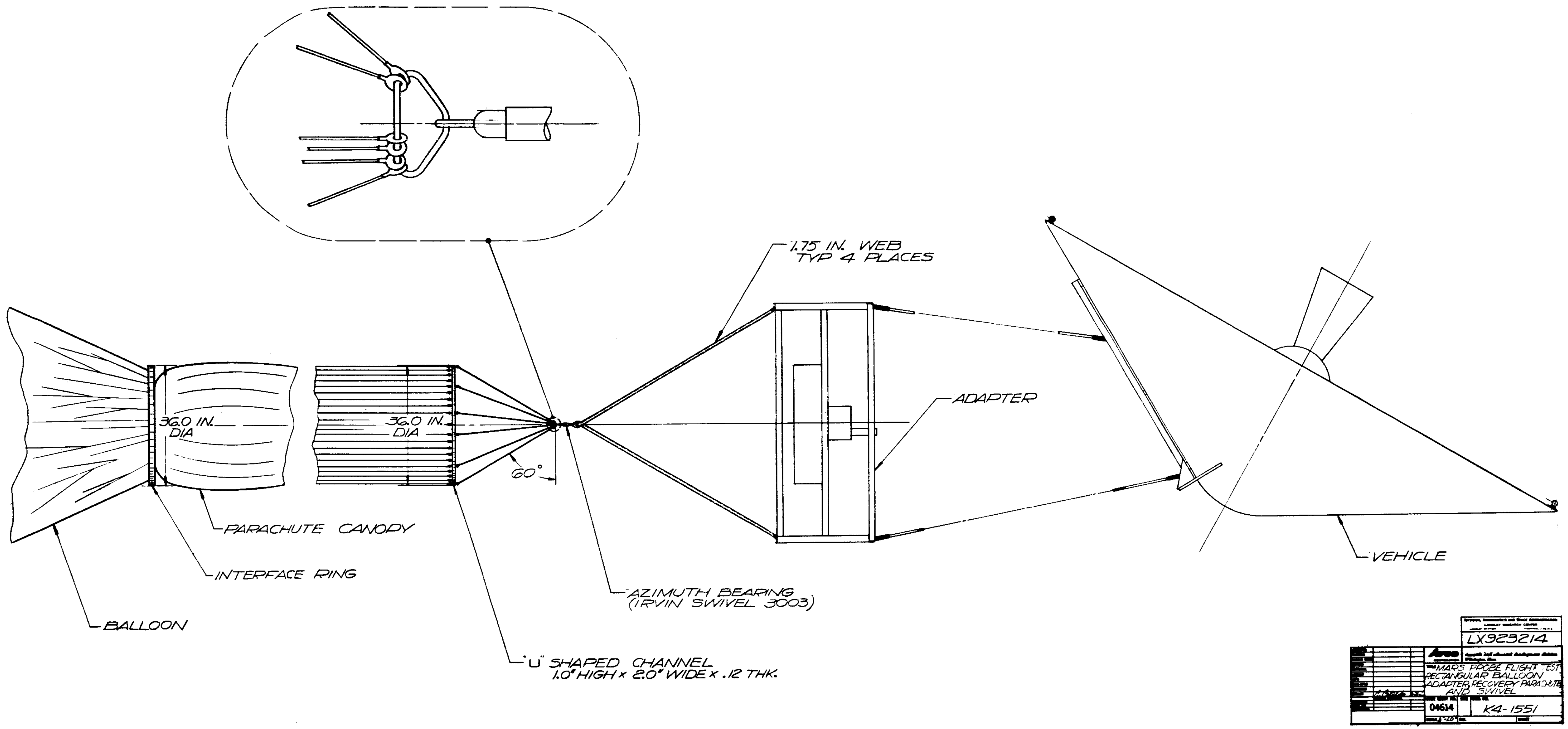

Figure 21. BALLOON ADAPTER CONFIGURATION -- FULL-SCALE PRE-VOYAGER PARACHUTE FLIGHT TESTS -- ROCKET CLIMB FROM BALLOON RELEASE

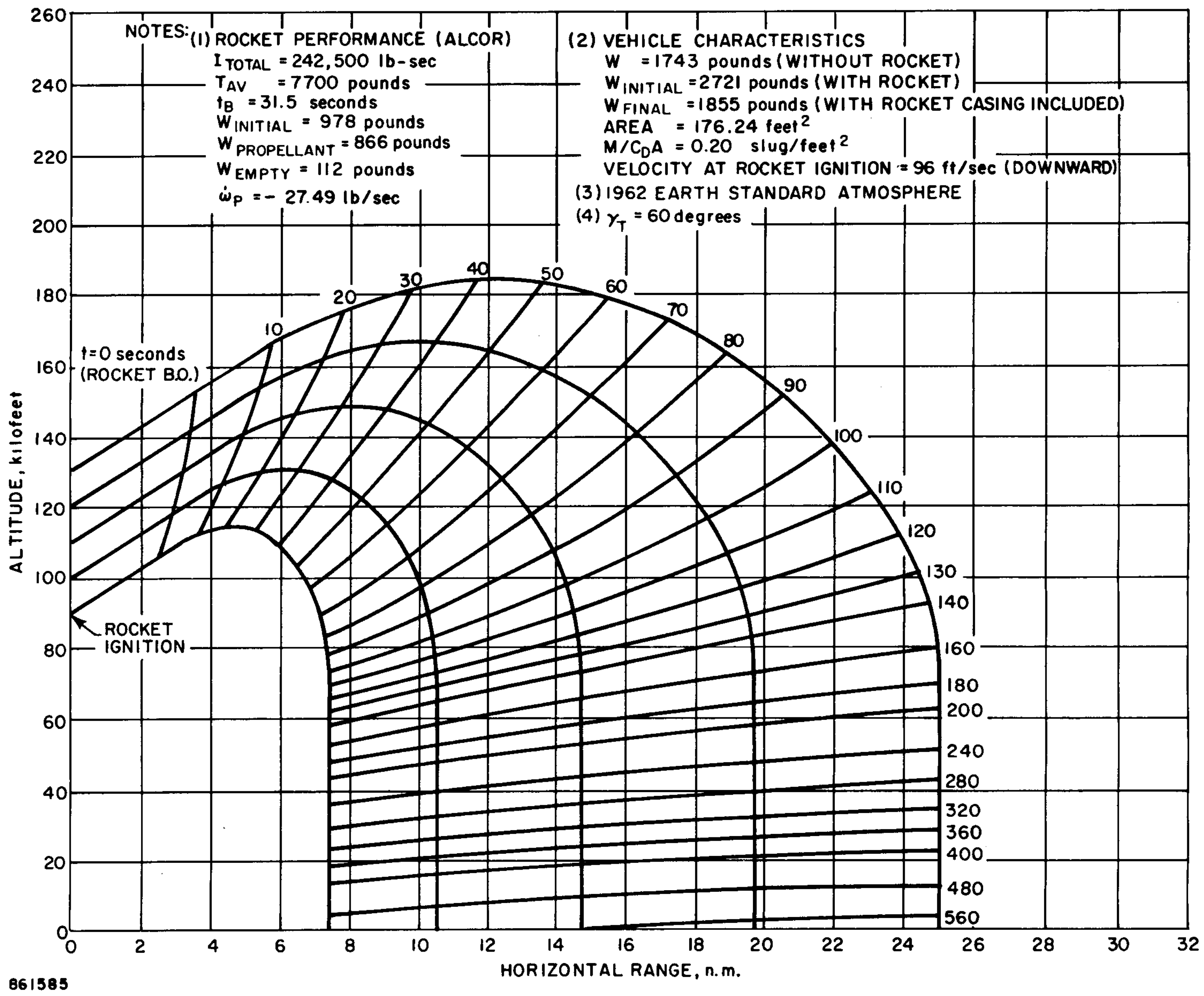

Figure 20 TEST VEHICLE HORIZONTAL RANGE - FULL-SCALE PRE-VOYAGER PARACHUTE FLIGHT TESTS - ROCKET CLIMB FROM BALLOON RELEASE

shroud lines through a mechanical bearing while azimuthal training is accomplished in the same manner. The recovery parachute canopy is attached to the balloon base with a large diameter ring.

8.3.3 Test Vehicle Configuration

Since this vehicle is designed for a pre-Voyager test program, and the ultimate operational design will not have been defined at that time, the attempts to simulate current operational concepts will be restricted to the following:

1. The external vehicle shape to check wake effects.

2. The separation of the entry-vehicle shell from the parachute suspended payload immediately after peak opening shock loads to check mass change effects on parachute performance and not separation system performance.

3. The mass of the vehicle shell

4. The mass and weight force of the suspended payload

The vehicle consists of three major subassemblies; the external shell, the internal structure (suspended payload), and the Alcor rocket with attached ballast as illustrated in the inboard profile of Figure 22. The vehicle may be further broken down into its significant subsystems and components:

1. External structure
2. Internal structure
3. FM/FM telemetry
4. Battery Power supply
5. Programmer and control circuitry
6. Main Parachute
7. Pilot Chute
8. Cameras
9. Tensiometer
10. Accelerometers
11. Rate gyros

12. Pitot-static tube

13. Alcor solid rocket

14. Spin rocket

15. Vehicle shell separation system

16. Rocket case separation system

17. Recovery parachutes (2)

18. Umbilical connector

19. Ballast

The blunt cone external shell is similar to the operational prototype shape conceived during this study in order to provide a reasonable simulation of blunt vehicle wake effects on the parachute. The dimensions are full size and the mass is the same as the operational prototype. It is a monocoque shell, stiffened against buckling by a skeleton frame of angles and a fabricated outer ring. No heat shield material is necessary since reentry or high-speed flight is not involved in the test. The spin rockets are mounted on the outer ring and the shell recovery parachute (if required for range safety) is installed on the inner surface of the shell. The internal structure (suspended payload) mates with the inside of the shell at a separation ring as shown in the inboard profile (Figure 22). The separation mechanism is a Marmon clamp. Bearing support between the shell and the Alcor rocket is provided for the rocket-thrust loads. The internal structure consists of a truss assembly which supports the Alcor rocket and other subsystems. The parachute harness is attached to the truss assembly. A swivel support is provided between the attachment harness and the parachute shroud lines since the vehicle is not despun. Assuming that the prototype spin dynamic analyses are applicable, a spin rate of about 30 to 40 rpm will be adequate and this rate will not adversely affect parachute performance if a swivel is used. The Alcor rocket is attached to the truss by four explosive nuts to facilitate separation for weight force simulation. Ballast and a recovery parachute (if required for range safety) are mounted on the rocket case.

Two cameras, aimed rearward, are mounted on the truss assembly for recording parachute deployment and descent performance at high-and low-frame speeds. A tensiometer between the parachute shroud lines and the attachment harness measures opening shock and drag loads. Other instrumentation consists of accelerometers, rate gyros and a pitot-static tube. All data except the camera film are telemetered to the ground.

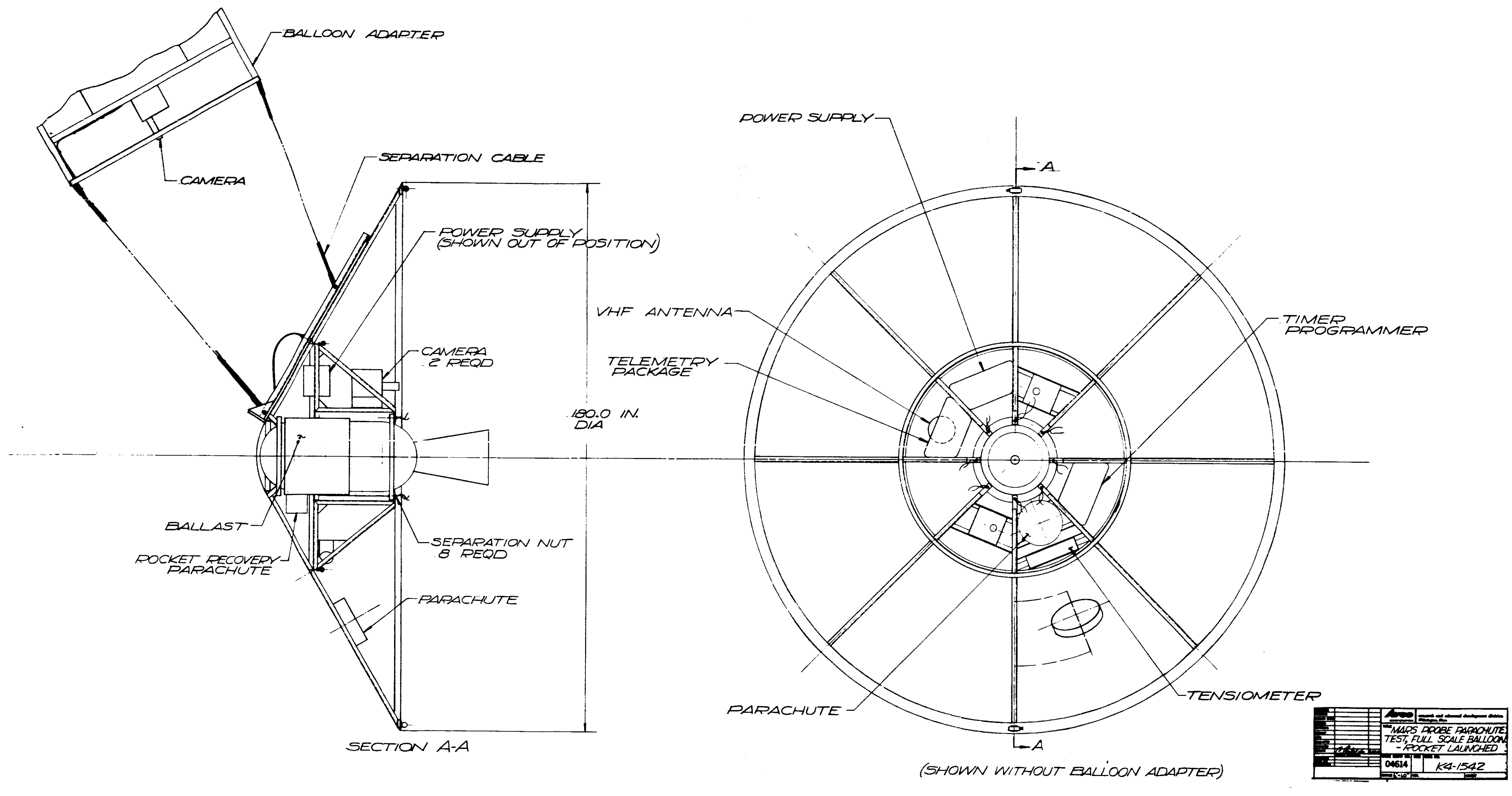

Figure 22 INBOARD PROFILE OF FULL SCALE PRE-VOYAGER PARACHUTE FLIGHT TEST VEHICLE -- ROCKET CLIMB FROM BALLOON RELEASE

135-1

8.3.4 Flight Sequence

The flight sequence of the balloon-launched full-scale parachute test is illustrated in Figure 23. The ballon is released some distance upwind of the the test range such that it will be near the ground tracking stations and safe recovery areas when the release altitude is reached. This is usually implemented by transporting a few days before launch, the uninflated balloon and test vehicle, to a predicted upwind launch point based on prevelant wind directions. Launching is delayed until the right combination of safe ground winds and desired winds aloft occur. Depending on the season of the year this delay can reach a week or more. After release, the ballon will ascend at a rate of about 1000 feet per minute. This ascent rate can be controlled by radio command signals to the gas valve and ballast control systems on the balloon adapter if winds aloft and ballon drift become unfavorable. The technique is to slow ascent in favorable wind strata and hasten ascent in unfavorable strata. During the ascent, which takes about 1-1/2 hours, the balloon is tracked by radar, cinephototheodolites, and aircraft, and altitude data are telemetered to the ground from the balloon.

As the release attitude, 110,000 feet, is approached, the test vehicle and balloon adapter are rotated to the desired launch azimuth by the azimuth control system. The desired launch azimuth is transmitted by radio command. The selected direction will be towards the test range. Although the horizontal range from test vehicle launching to parachute deployment is not excessive (5 nm) the control of launch azimuth will ease range safety requirements on acceptable winds aloft.

Just prior to release, the vehicle systems are energized and warmed up by the balloon adapter battery. The vehicle telemetry is calibrated, the electrical system is switched to internal power, the balloon camera is turned on and the vehicle released by ground command.

At release, a lanyard from the balloon adapter triggers the programmer, which controls the remaining vehicle sequences. The spin rockets are fired immediately which stabilizes the vehicle at its release attitude of 60 degrees. The Alcor rocket is ignited at spin rocket burnout and the vehicle starts climbing and accelerating in velocity. Alcor burnout occurs 31.5 seconds after ignition at an altitude of 130,000 feet and a velocity of 1750 ft/sec. The vehicle then coasts in a climb with the velocity slowly decreasing under gravity and drag forces. About 9 seconds after burnout the vehicle reaches an altitude of 140,000 feet at a velocity of 1250 ft/sec which is close to the high-energy end of the operational deployment envelope ($q = 4$ lb/ft^2 and $M = 1.2$).

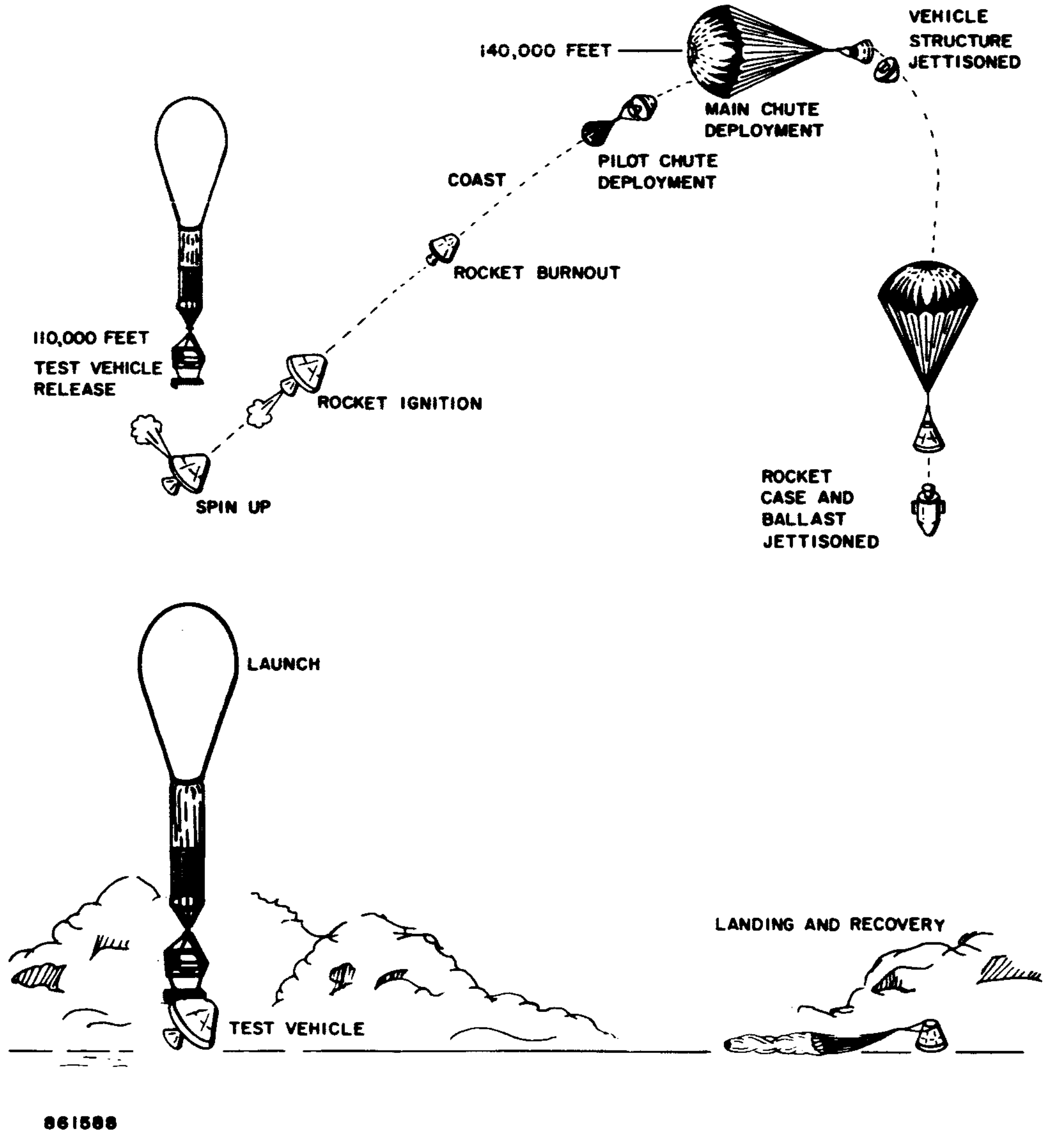

Figure 23 FULL-SCALE PRE-VOYAGER PARACHUTE TEST - ROCKET CLIMB FROM BALLOON RELEASE - FLIGHT SEQUENCE

The programmer (or radio command or other internal instrument, if required, for reducing deployment dispersion) starts the parachute cameras and triggers mortar ejection of the pilot chute, which in turn pulls out the main parachute. Shortly after the peak opening shock load as indicated by an accelerometer, the vehicle shell is jettisoned. Terminal velocity is reached in a few seconds. After the parachute and suspended payload descend for about 30 seconds, the expended Alcor rocket case and attached ballast are jettisoned to reduce the payload weight to a value equal to the Mars weight force. Thirty seconds later the parachute and payload have descended to about 120,000 feet at which point the Earth's atmospheric density is equal to the Mars surface density for the minimum atmospheric model VM-7. Below this altitude the environmental simulation degenerates as the density increases and the descent velocity decreases below operational values. The descending parachute and payload are tracked to the ground for recovery of the camera and parachute for postflight evaluation.

The test vehicle shell, which was jettisoned after parachute deployment, may be recovered by parachute, solely for range safety purposes. There is a tradeoff here in that recovery by parachute will reduce impact velocity, but will increase wind drift dispersion and its attendant hazards.

The balloon adapter is also recovered by parachute. After test vehicle release, a radio command releases the recovery parachute canopy from the balloon.

In one of the two recommended full-scale tests the entry vehicle shell separation will not occur at the peak opening shock load, but will be delayed for about 20 seconds. The purpose is to allow more time for measurement of possible blunt-body wake effects on parachute performance. The 20-second delay is a significant deviation from the operational mode but is necessary since the operational mode permits only a transient application of the wake and marginal effects such as incipient inflation failures may be random enough not to be detected in one or two tests.

8.4 VOYAGER SUBSCALE PARACHUTE TESTS

8.4.1 Test Program

The requirements and environment for Voyager subscale tests are identical to the pre-Voyager subscale tests hence the same test program is recommended: Nike/Nike/Dart in one-tenth scale evaluation. Fewer flights are required, however, since only one parachute configuration will be evaluated. Ten tests are recommended. The discussion of the Nike/Nike/Dart selection rationale, launch vehicle configuration, test vehicle configuration and flight sequence for the pre-Voyager program presented in Section 8.2 is applicable here and will not be repeated.

The test program consists of ten flights at six deployment conditions for both payload mass and payload weight simulation. Four of the deployment conditions will be at the extremities of the operational deployment envelope for both mass and weight simulation. Two flights with only payload mass simulated will be made at dynamic pressures slightly higher than the operational envelope at both minimum and maximum Mach number. These two flights are dynamic structural tests and hence require only payload mass simulation since the weight simulation will produce smaller loadings on the parachute.

8.5 VOYAGER FULL SCALE PARACHUTE/SEPARATION TEST

8.5.1 Test Program

Flight tests of the full-scale parachute will be combined with the separation subsystems tests in the Voyager program. The launch vehicle selected for the program is the Little Joe II. Details of this program will be described in the Separation Section 9.0. Only details of the parachute program objectives will be discussed here.

The test program consists of ten flights at seven deployment conditions for both payload mass and payload weight simulation. Five of the development conditions will be at the extremities of the operational envelope for both mass and weight simulation. Two flights with only payload mass simulated will be made at dynamic pressures slightly higher than the operational envelope at minimum and maximum Mach number. These two flights are dynamic structrual tests and hence require only payload mass simulation. The flights made at high Mach numbers in the operational envelope will include both payload mass and payload weight simulation on each flight. The low Mach number flights which occur at lower altitudes will be restricted to single simulations, either mass or weight, because of limited descent time for accommodating both simulations. The limited descent time refers to the time between deployment and the altitude at which the Earth's atmospheric density becomes greater than the Mars surface atmospheric density. This altitude is a little under 120,000 feet for the minimum surface density model, VM-7. Below this altitude, the increased density reduces descent velocity and produces flow conditions which deviate from operational values.

9.0 SEPARATION SYSTEMS FLIGHT TESTS

9.1 TEST REQUIREMENTS AND OBJECTIVES

The status of vehicle separation technology was judged adequate for the requirements of the Voyager program and no pre-Voyager technological development testing, either ground or flight, is recommended. Development flight testing for design verification during the Voyager program is recommended. Combining separation testing and full-scale parachute testing on the same flights is recommended because of their compatibility and the attendent cost savings.

The technology of separation in terms of release mechanisms, ejection force mechanisms, minimizing mechanical interface interference, reliability of mechanical and electrical interfaces, etc., has a significant history of development and flight experience with hardware available in shelf item or near shelf item status. These techniques and hardware are applicable to the Voyager requirements and hence technological development is not necessary. Program development flight testing, however, was judged necessary because of the large number of separation functions, the complexity of some of the separations and limitations in ground test physical simulation. The separation of the sterilization canister lid, the capsule from the spacecraft, and the entry-vehicle shell from the suspended payload are complex separations involving large structures with mechanical interfaces of large dimension. The entry-vehicle shell separation, in particular, occurs in a dynamic environment with aerodynamic loads on the shell, large parachute opening shock loads on the suspended payload, and with the vehicle possibly spinning and oscillating in angle of attack. Adequate simulation of this environment in ground tests is not feasible. Despite the simulation inaccuracies, ground tests are still recommended because of the opportunity for vastly superior instrumentation and visual observation. It should also be noted that ground test simulation can be very good for vacuum flight separations such as the canister lid when tested with ballistic pendulum techniques.

Incorporation of the separation tests with the parachute flight tests was a logical choice since all but two of the separations (canister lid and capsule/spacecraft) occur as part of parachute deployment or during parachute descent. These separations are pilot chute, main parachute, entry vehicle shell and penetrometers. It should be noted that the addition of parachute testing to the separation flights resulted in a compromised environmental simulation for the two vacuum flight separations (canister lid and capsule/spacecraft). Trajectory analysis indicated that the apogee of the trajectory must be restricted to a maximum of 170,000 feet or less and the velocity at apogee must be low. In descending from this apogee and low velocity the vehicle will accelerate to the correct velocity at the deployment altitude. Descent from higher apogees will yield velocities beyond the deployment envelope. An altitude of 170,000 feet is within the sensible atmosphere, and although the low velocities are providing very low dynamic pressures ($q = 0.1$ lb/ft^2), some aerodynamic loading exists on the separated hardware and vacuum flight is not truly simulated. It is felt that the loading is not large enough to invalidate the test. The only alternative is to schedule independent flight tests of the vacuum flight separations, and accept increased program costs.

9.2 TEST PROGRAM

The recommended program for the combined separation and parachute full-scale flight tests is the surface launched Little Joe II providing a high altitude trajectory for a full-scale boilerplate mockup of the entry vehicle and its sterilization canister. Vacuum flight separations are tested at apogee (170, 000 feet) and parachute deployment and subsequent separations occur during descent from apogee. The Little Joe II is a versatile vehicle which utilizes for main propulsion, clusters of Algol solid rocket motors in various staging combinations up to a total of seven Algol motors. Flight path control is provided by an autopilot driving aerodynamic fins and reaction gas jets. The test vehicle is a boilerplate mockup in which the mass characteristics and external configuration of all separation subassemblies duplicate operational configurations. Separation mechanisms and the parachute test conditions and repetitive checks of the separation functions, are recommended for the program. Test conditions for the ten flights are described in the Voyager full-scale parachute section (paragraph 8.5.1).

As in the case of the pre-Voyager full-scale parachute tests, the large dimensions of the test vehicle reduce the potential launch techniques to the two choices: surface launched Little Joe II or rocket climb after balloon release. Unlike the pre-Voyager test, the Little Joe II was selected in preference to the balloon approach because of the increased complexity of the balloon test vehicle. This complexity was due to the trajectory requirements which eliminated vehicle spin stabilization for flight path control (and TVC) and necessitated an active, closed-loop flight control system consisting of an autopilot and outboard reaction motors. As explained in the previous section, a trajectory apogee of 170, 000 feet or less at low velocity is necessary. The balloon launched test vehicle must climb at a very steep angle to achieve these apogee conditions, but the climb angle is limited, due to the presence of the large diameter balloon above the vehicle. The solution is to use a programmed climb which begins at acceptable climb angles and becomes steeper after the balloon is cleared. The programmed climb eliminates spin stabilization and requires the active control system. The actively controlled vehicle represents a significant increase in complexity, which will require development and flight testing schedules not compatible with the Voyager flight test program schedules. The decision, therefore, went to the existing vehicle.

9.3 LAUNCH VEHICLE CONFIGURATION

The significant features of the Little Joe II configuration required for this test are 2 and 1 Algol rocket staging, controllable fins, 198 inch hammerhead ascent shroud, four retrothrust Recruit rockets, and the pitch-roll gyro replaced with rate gyro integration.

The first-stage burn with two Algol rockets and second-stage burn with one Algol require Algol 2B motors for tests to the high-energy end of the deployment envelope, and Algol 1D motors for tests to the low-energy end. Another possibility is to employ Algol 2B motors for the low-energy deployment and add extra ballast. The controllable (not fixed) fin assemblies must be employed both for ascent flight path control and pitch attitude maneuvers near apogee. The controllable fin assemblies consist of not only the servo controlled aerodynamic fin but also the reaction jet system used for the pitch attitude maneuvers.

The test vehicle diameter of 15 feet is greater than the basic 13-foot diameter of the Little Joe II, and a hammerhead ascent shroud will be required. The maximum diameter will be 198 inches. Experience with hammerhead shrouds on the Little Joe II is limited to wind-tunnel tests of a 212-inch shroud for the LEM. These tests showed no loss in fin effectiveness. Wind-tunnel tests, and design and development of the specific shroud configuration will be required.

No retrothrust capability exists in the current versions of the Little Joe II. Preliminary analysis revealed a retrothrust requirement which could be supplied by four Recruit motors. These would be installed near the base of the vehicle and inclined to the vehicle centerline such that they could exhaust through holes cut in the side of the vehicle. Thrust would be rearward. There is plenty of room for this installation and the structural changes are feasible according to the contractor. The four Recruits will provide a 4-g deceleration for 1.5 seconds resulting in a velocity decrement of 200 ft/sec. This large velocity decrement is necessary since the parachute deployment occurs a relatively short time later (10 to 20 seconds) and the possibility of a collision must be minimized.

The two large pitch attitude maneuvers (about 180 degrees) will result in unacceptable cross coupling in the pitch-roll gyro. This problem is easily solved by integrating the rate gyro outputs to get attitude data for the autopilot.

No test vehicle separation system need be designed for the test vehicle/Little Joe II adapter. The operational spacecraft/capsule separation system will be used since its evaluation is one purpose of the test. The canister afterbody and spacecraft/capsule adapter will remain attached to the vehicle. Hardened separation cameras, mounted on the canister afterbody, must be recovered after the Little Joe II impact.

9.4 TEST VEHICLE CONFIGURATION

The major subassemblies of this test vehicle will be mockups of the operational prototype. The subsystems being tested (viz., parachute and separation systems) will be operational prototypes or as close to operational prototypes as the status of the design and development permits. The major subassemblies are the sterilization canister lid and afterbody, entry vehicle shell, suspended capsule, spacecraft/capsule adapter and test vehicle/Little Joe II adapter. An inboard profile of the test vehicle mounted within the ascent shroud of the Little Joe II launch vehicle is shown in Figure 24.

The first-stage burn with two Algol rockets and second-stage burn with one Algol require Algol 2B motors for tests to the high-energy end of the deployment envelope, and Algol 1D motors for tests to the low-energy end. Another possibility is to employ Algol 2B motors for the low-energy deployment and add extra ballast. The controllable (not fixed) fin assemblies must be employed both for ascent flight path control and pitch attitude maneuvers near apogee. The controllable fin assemblies consist of not only the servo controlled aerodynamic fin but also the reaction jet system used for the pitch attitude maneuvers.

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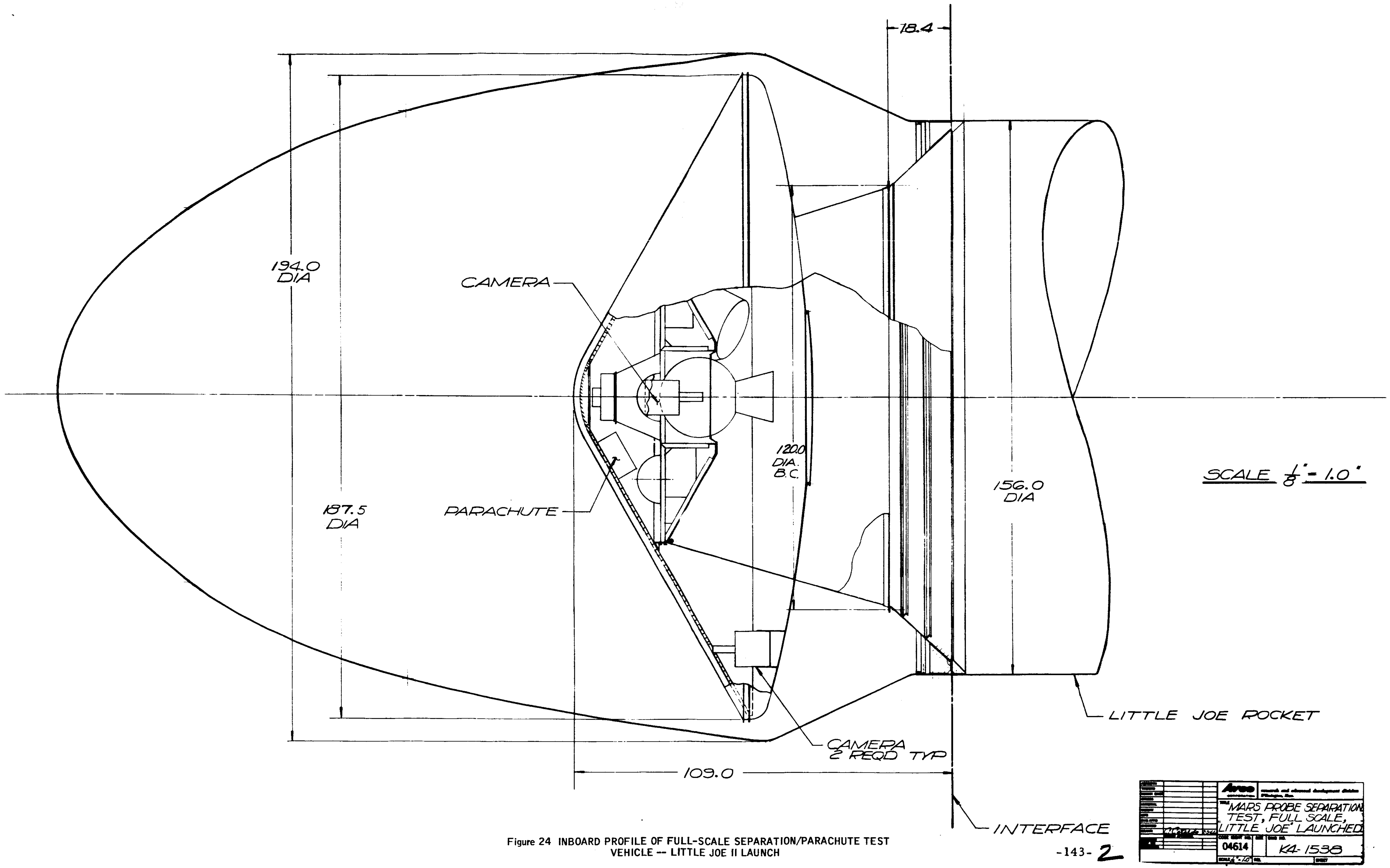

Figure 24 INBOARD PROFILE OF FULL-SCALE SEPARATION/PARACHUTE TEST VEHICLE -- LITTLE JOE II LAUNCH

143.1

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al dynamics at separa-
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gram and hence will
m. The shell mockup
The mockup will be
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atching the mass char-
n the test vehicle and
The suspended capsule
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cameras are located
hicle as shown in

The significant subassemblies and components of the vehicle are summarized as follows:

1. Sterilization Canister Lid
2. Sterilization Canister Afterbody
3. Test Vehicle External Shell
4. Internal Structure (or Suspended Capsule)
5. Spacecraft/Capsule Adapter
6. Test Vehicle/Little Joe II Adapter
7. FM/FM Telemetry
8. Battery Power Supply
9. Programmer and Control Circuitry
10. Main Parachute
11. Pilot Chute
12. Parachute and Separation Cameras
13. Separation Sensors
14. Tensiometer
15. Accelerometers
16. Rate Gyros
17. Ram and Static Pressure Sensors
18. External Shell Recovery Parachute
19. Umbilical Connector
20. De-orbit Rocket Propellant Ballast
21. Weight Force Simulation Ballast

The sterilization canister lid and afterbody and spacecraft/capsule adapter can be boilerplate mockups or prototype structure if the prototypes are available at this time in the program. Mass and moment of inertia of the canister lid must duplicate prototype values to properly simulate operational dynamics at separation. The entry-vehicle shell, which is a lightweight honeycomb of very large dimension, will require a manufacturing development program and hence will probably not be available in time for the flight test program. The shell mockup must match the prototype in mass and moment of inertia. The mockup will be a ring-stiffened monocoque shell which is inherently a heavier fabrication than the prototype honeycomb. No problem is anticipated in matching the mass characteristics, however, because no heat shield is required on the test vehicle and the weight saved can be accommodated in the structure. The suspended capsule (or internal structure) must not only match prototype mass characteristics but its external configuration, including small protuberances (if any), must be carefully duplicated. This is necessary to properly test possible interference and fouling between the parachute attachment harness and adjacent structure during the deployment sequence. Similar care must be exercised in the detailed configuration of other subassembly mockups, such as the canister afterbody, to ensure that the test of subassembly separations will be a valid check of possible interferences. The de-orbit rocket nozzle which protrudes from the suspended capsule in the vicinity of the parachute harness, must also be simulated for this reason.

Although the inboard profile shows a complete de-orbit rocket case, only the nozzle and that part of the case that protrudes need be duplicated. In fact, it may be necessary to minimize the rocket mockup to save weight, in order to accommodate the weight force simulation. This may also require weight economies in the design and fabrication of the suspended capsule structure. To illustrate this point, the prototype weight of the suspended capsule (not including the parachute) is 940 pounds. For the weight force simulation, this weight must be reduced to 39 percent, or 367 pounds. The prototype structural weight is 150 pounds, the expended rocket case is 49 pounds, and the estimated flight test telemetry instrumentation, power supply, etc. weight is 125 pounds. The total is 324 pounds, which provides a margin of 43 pounds or 12 percent, a rather small margin for growth. Shot or dust ballast weighing 573 pounds is ejected from the suspended capsule during the parachute descent to reduce the capsule weight to the 367 pounds required for the weight force simulation. Additional ballast weighing 420 pounds is jettisoned after vehicle separation from the Little Joe II, but before parachute deployment. This ballast compensates for the propellant mass that would have been expended by the de-orbit rocket and TVC system. The vehicle is free-falling and accelerating (by gravity) when this ballast is jettisoned and hence the shot or dust must be forcibly ejected by mortar, mechanical spring, or other suitable system.

Separation performance is recorded by cameras and separation sensors such as spring-probe or lanyard-spool devices. Canister lid separation is photographed by two cameras mounted on the canister afterbody which view the lid through observation parts in the entry vehicle shell. The cameras are located near the maximum diameter and at opposite sides of the vehicle as shown in

Figure 24. These cameras also record separation of the test vehicle from the Little Joe II. The cameras must be hardened to survive the Little Joe II ground impact. Another camera mounted on the suspended capsule photographs the separation of the entry vehicle shell. Two other cameras also mounted on the suspended capsule but facing rearward monitor parachute performance during deployment and descent.

A parachute mounted on the inside of the entry vehicle shell is used for shell recovery for range safety. There is a tradeoff here, in that the parachute will reduce impact velocity but also increase wind drift magnitudes. The latter could be a deleterious factor in launch aborts due to wind drift/range safety conflicts.

9.5 FLIGHT SEQUENCE

The flight sequence for the parachute/separation test is illustrated in Figure 25. The Little Joe II is launched at a pitch attitude of 84 degrees, by ignition of the two first-stage Algol 2B rockets. Second-stage ignition of the single Algol 2B motor occurs at 41, 000 feet at t + 60 seconds. Second-stage burnout occurs at 128, 000 feet at t + 126.4 seconds. The ascent shroud is jettisoned at 152, 000 feet at t + 145 seconds. Shortly after shroud jettison, the vehicle pitch attitude is reversed and stabilized at this attitude by the reaction gas jet system. Apogee occurs at 170, 000 feet at t + 175 seconds where the canister lid is jettisoned. The pitch attitude is again reversed and stabilized by the reaction jet system. The entry vehicle is then separated from the canister afterbody which remains attached to the booster. retrorockets are fired and the entry vehicle descends at increasing velocity towards the deployment altitude. Before reaching the deployment altitude, ballast which simulates the mass of the de-orbit rocket propellant is jettisoned. The pilot and main parachutes are deployed at 140, 000 feet where the vehicle velocity will be about 1270 ft/sec (M = 1.2) and the dynamic pressure will be 4 lb/ft^2. The entry-vehicle shell is separated at the peak opening shock load. This shell may or may not be recovered by parachute depending upon range safety requirements. During the parachute descent, the penetrometers are released. The vehicle is recovered after impact for postflight examination of the parachute and separation systems.

9.6 ALTERNATIVE TEST METHOD CONSIDERED - BALLOON LAUNCHED FULL SCALE TEST VEHICLE

9.6.1 Test Program

Although not chosen as the recommended test method, the program is described because of possible interest in an unusual test technique.

This alternative program for the full-scale tests of the parachute and separation systems consist of a rocket-propelled climb of the test vehicle after balloon release at high altitude. Some separation systems are tested at

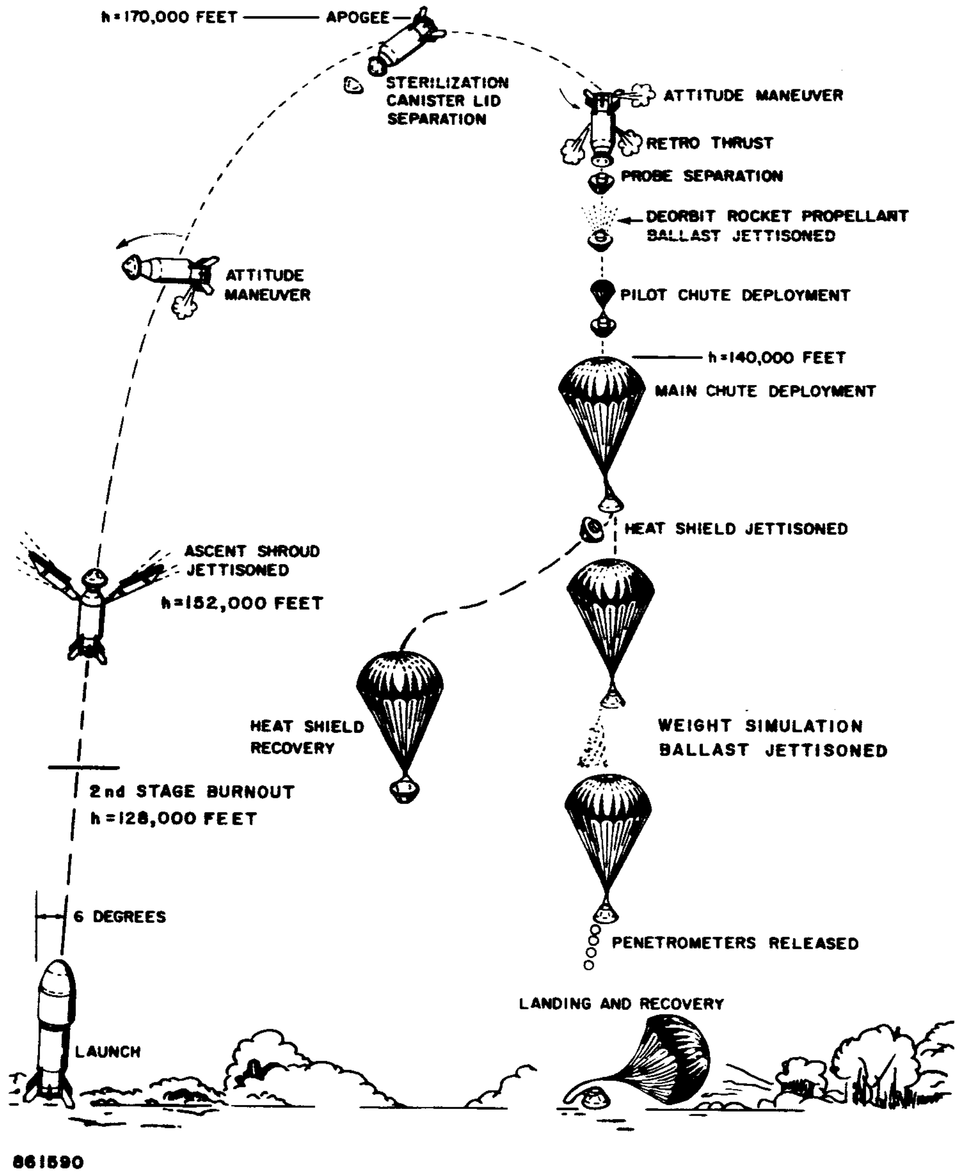

Figure 25 FULL-SCALE PARACHUTE/SEPARATION TEST - LITTLE JOE II LAUNCH-FLIGHT SEQUENCE

apogee and the rest during and after the parachute deployment which occurs during descent from apogee. The test vehicle is a boilerplate mockup of both the prototype entry vehicle and the sterilization canister. A Surveyor rocket and two Ranger rockets are used to propel the vehicle in a programmed climb under control of an active, closed-loop flight control system. The balloon is a zero pressure-type fabricated of Mylar film reinforced with bonded Dacron scrim. Balloon sizes between 4 and 20 million cubic feet will be required depending on the desired test conditions.

The performance of the vehicle is demonstrated in Figure 26, which consists of the vehicle trajectories plotted in coordinates of altitude versus velocity for various launch altitudes. Two-stage rocket firing is used for the powered climb. The two Ranger rockets are fired first with the vehicle stabilized at an attitude angle of 60 degrees (nose-up). Ranger rocket burnout, which is indicated on the trajectories, is followed by a 5-second ascent coast during which the vehicle attitude is pitched up to 90 degrees and stabilized at this attitude during the Surveyor burn. The vehicle decelerates during the 5-second coast due to drag and gravity as shown by the curves. A major part of the velocity and altitude increase is provided by the Surveyor rocket as is evident. After Surveyor rocket burnout, the vehicle ascends towards apogee where the velocity has dropped to low values (about 200 ft/sec). At this point the dynamic pressure is very low, as indicated by the contour line of constant dynamic pressure ($q = 0.1$ lb/ft^2). Note that all the apogee occur at dynamic pressures of $q = 0.1$ lb/ft^2 or less regardless of launch altitude. This is important for initiation of the vacuum flight separations at apogee. As the vehicle descends from apogee, velocity increases under gravitational acceleration and the trajectory passes through the deployment envelope as shown in Figure 26. The test condition attained within the envelope depends on the apogee altitude which in turn is a function of the launch altitude. As indicated in the figure, launch altitudes between 90,000 and 120,000 feet will provide complete coverage of the operational envelope. Discrete time marks on the trajectories indicate that the time interval between apogee and deployment varies between 25 and 40 seconds. This duration should be adequate for the separation functions and attitude maneuvers which must be completed during this interval.

The trajectories in Figure 26 were computed before the vehicle was designed, and it was assumed that the thrust lines of the Surveyor and Ranger rockets were aligned parallel to the vehicle centerline. During the configuration layout it was necessary to align the Ranger rocket thrust lines 30 degrees to the vehicle centerline. This will reduce the Ranger rocket total impulse by 13.4 percent. Another trajectory was computed (Figure 27) to check the effect of this total impulse reduction. Comparing Figure 26 with Figure 27 indicates that the effect is equivalent to a reduction of 5,000 feet in the launch altitude, which is not significant. Figure 26 can be used for reference purposes by simply accounting for this launch altitude correction.

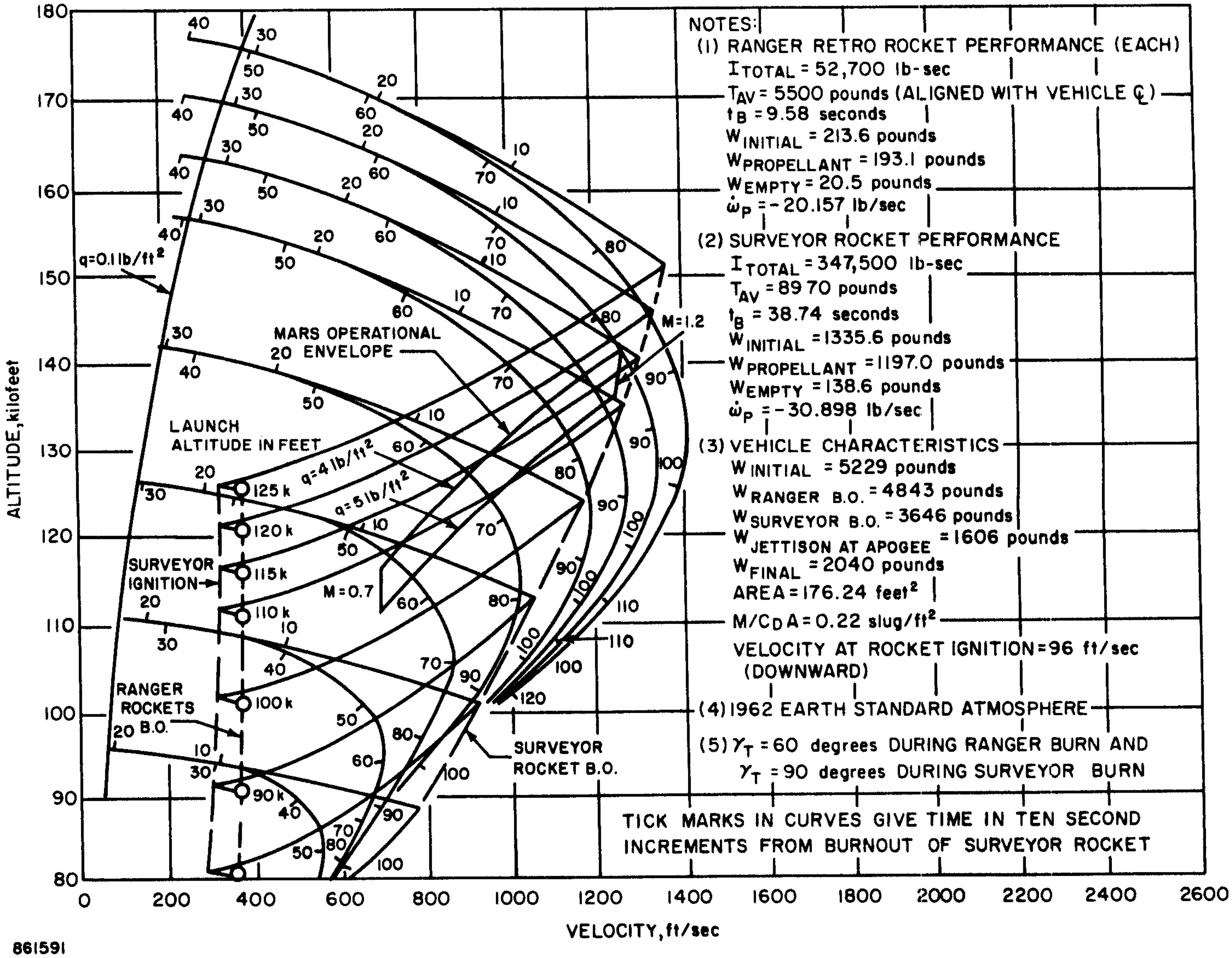

Figure 26 TEST VEHICLE TRAJECTORIES - FULL-SCALE SEPARATION/PARACHUTE TEST - ROCKET CLIMB FROM BALLOON RELEASE

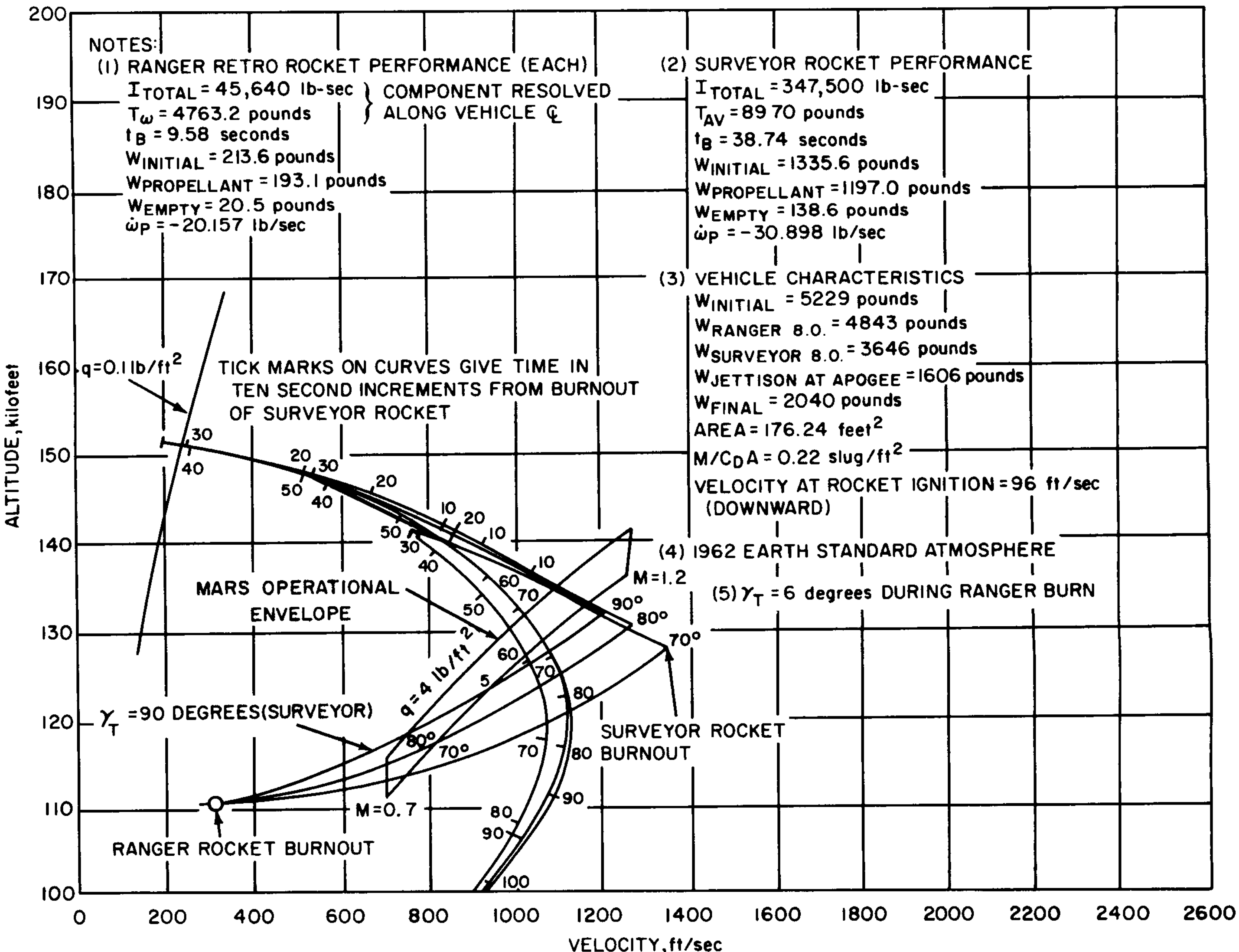

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Figure 27 TEST VEHICLE TRAJECTORIES - FULL-SCALE SEPARATION/PARACHUTE TEST - ROCKET CLIMB FROM BALLOON RELEASE - 30-DEGREE RANGER ROCKET THRUST LINE

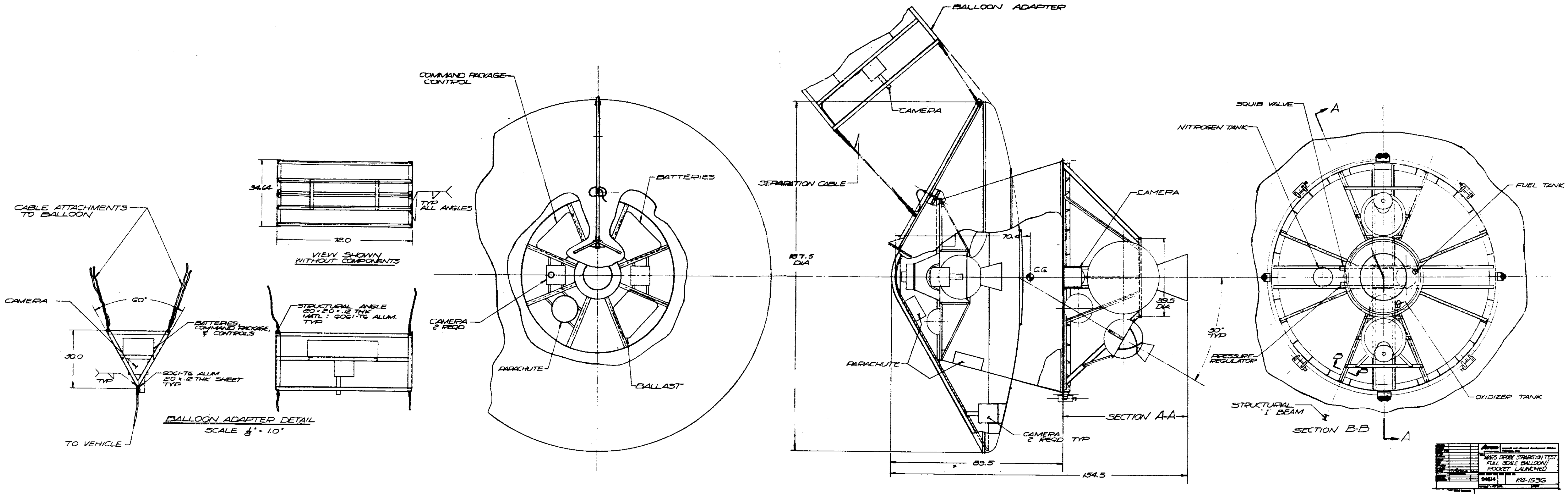

Figure 28 INBOARD PROFILE OF FULL-SCALE PARACHUTE/SEPARATION FLIGHT TEST VEHICLE -- ROCKET CLIMB FROM BALLOON RELEASE

9.6.2 Balloon Configuration

The balloon configuration consists of a reinforced Mylar, zero-pressure balloon, recovery parachute and balloon adapter. Except for balloon size the configuration is identical to the balloon recommended for the pre-Voyager full-scale parachute tests which is described in Paragraph 8.3.2. The launch altitudes for this application vary between 90,000 and 120,000 feet which requires balloon sizes from 4 to 20 million cubic feet. The total balloon payload is 5500 pounds, 5200 pounds for the test vehicle and 300 pounds for the balloon adapter, which does not include the ascent control ballast which would have been jettisoned before the launch altitudes are reached.

9.6.3 Test Vehicle Configuration

The test vehicle configuration of this alternate approach for full-scale parachute/separation flight testing is similar to the configuration used with the Little Joe II launch vehicle. Forward of the spacecraft/capsule adapter the only differences are the addition of the recovery parachute for the canister afterbody and mounting fittings for suspension from the balloon adapter. This part of the configuration will not be discussed, as the description in Section 9.4 is applicable. Aft of the spacecraft/capsule adapter a propulsion package has been added. This consists of a Surveyor rocket, two Ranger rockets, a hot-gas TVC system with outboard nozzles, inertial reference system, autopilot, power supply, and supporting structure. An inboard profile for this configuration is presented in Figure 28.

The supporting structure is a tubular and I-beam truss, welded at one end to a stiffened vehicle interface ring and at the other to a mounting on the Surveyor rocket case. The truss was chosen over a monocoque structure because of the concentrated side loading from the two Ranger rockets. The Ranger rockets are installed in a cradle mounted between adjacent truss members. The Ranger rockets are installed such that their thrust line intersects the midpoint of the c.g. travel. The interface ring consists of welded flat plates and stiffeners. The ring is bolted to the spacecraft/capsule adapter.

The thrust vector control system is a liquid, bipropellant, hot-gas system with outboard nozzles mounted on interface ring. Pitch, yaw, and roll nozzle thrust requirements are 230, 115, and 30 pounds, respectively. Larger thrust values are required in pitch than in yaw in order to expedite the large pitch maneuvers. The roll nozzles are fired in couples, but not the pitch and yaw nozzles. Two 115-pound nozzles are used in pitch in order to achieve the required thrust value of 230 pounds with a nozzle currently available. The autopilot and its inertial reference and power supply are mounted on the truss. The TVC system is used in four modes: TVC during the Ranger rocket burn, pitch-attitude maneuver during the 5-second coast between staging, TVC during Surveyor rocket burn, and two 180-degree pitch-attitude maneuvers after burnout. The inertial reference will be established on the ground prior to balloon launch.

9.6.4 Flight Sequence

The balloon sequences of pre-launch, launch, ascent and test vehicle release are identical to the full-scale parachute test described in Paragraph 8.3.4. The sequences subsequent to vehicle release are entirely different and are described below. The flight sequence is illustrated in Figure 29.

The vehicle is released from the balloon at different altitudes to achieve different parachute deployment conditions. As a typical case, release at an altitude of 110,000 feet will be discussed. This will provide the deployment condition: $M = 1.0$, $q = 4.5$ lb/ft^2. The vehicle is released in a 60-degree climb attitude and the TVC is immediately activated. The two Ranger rockets are ignited 3 seconds after release. During burn the TVC system stabilizes the vehicle at the 60-degree attitude and the velocity increases to 310 ft/sec at burnout 9.6 seconds later. The gain in altitude is negligible since the vertical velocity at ignition was 100 ft/sec downward (due to 3 second free-fall), total velocity increase is small, and the burn time is short. The vehicle coasts for 5 seconds during which time the vehicle attitude is rotated to 90 degrees nose-up. The Surveyor rocket is ignited and the vehicle climbs at an increasing flight path angle which reaches 78 degrees at burnout, 38.7 seconds later. The altitude at burnout is 132,000 feet and the velocity is 1200 ft/sec. As the vehicle coasts upwards, its attitude is changed by the TVC system such that it will be flying backwards at apogee where the sterilization canister lid is jettisoned. Apogee is reached 35 seconds after burnout at an altitude of 151,000 feet and a velocity of 200 ft/sec. The dynamic pressure is less than 0.1 lb/ft^2 so the aerodynamic loads on the lid at separation are small. After lid separation the vehicle attitude is changed to the forward direction by the TVC system and the entry vehicle is separated from the canister afterbody, and propulsion structure.

Accelerating while free-falling from apogee, the vehicle reaches the deployment envelope at 130,000 feet, $M = 1.0$ and $q = 4.5$ lb/ft^2, for this particular case. The deployment envelopment is reached about 37 seconds after apogee. The subsequent deployment and separation sequences are identical to the Little Joe II parachute/separation test described in Section 9.5.

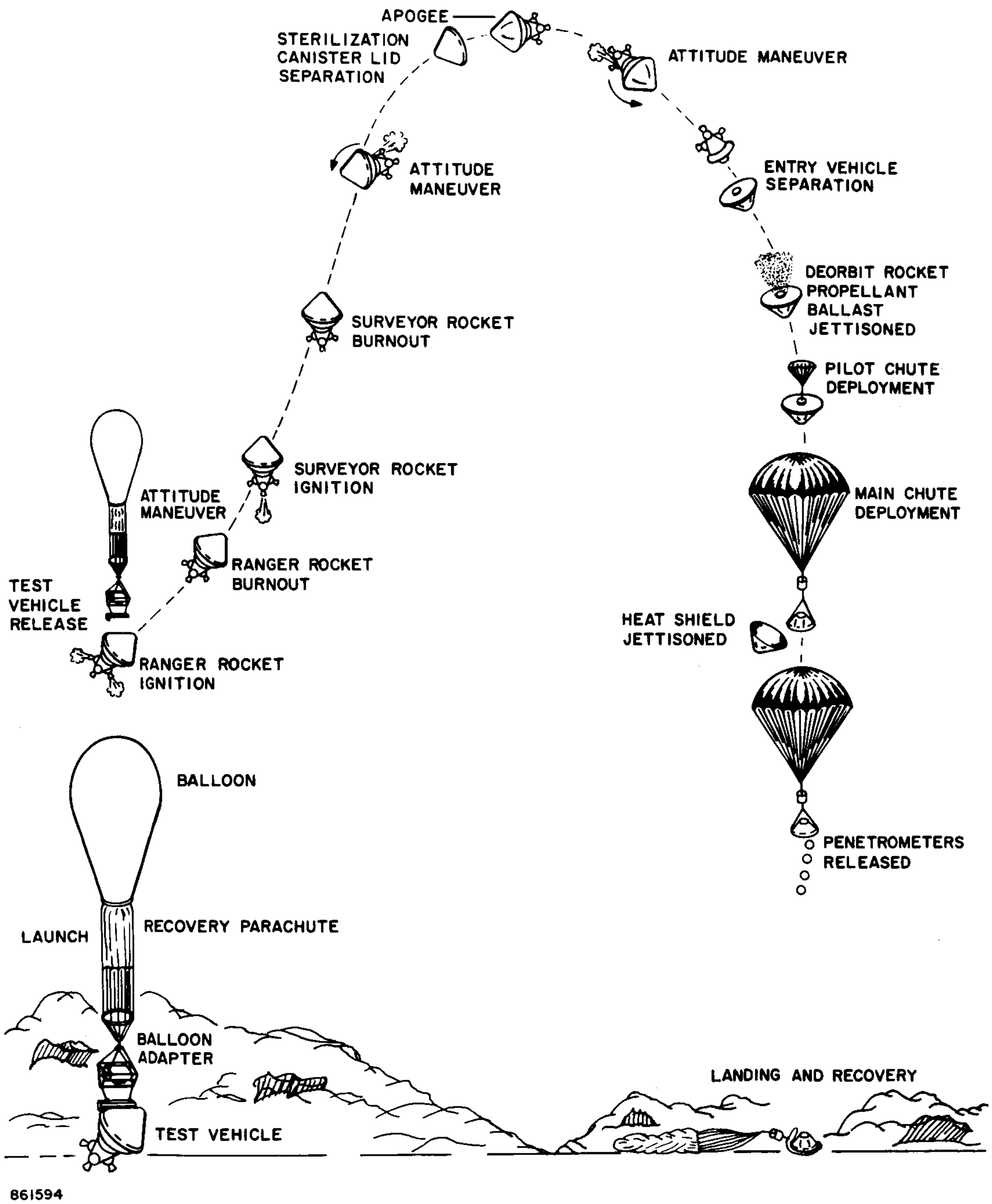

Figure 29 FULL-SCALE PARACHUTE/SEPARATION FLIGHT TEST - ROCKET CLIMB FROM BALLOON RELEASE - FLIGHT SEQUENCE

10.0 HEAT SHIELD PERFORMANCE FLIGHT TESTS

10.1 TEST REQUIREMENTS AND OBJECTIVES

10.1.1 General Considerations

Analytical performance prediction techniques which are based on ground test evaluation of the ablation materials are used in the design of the heat shield system. The ground test evaluation is subject to a number of limitations and the degree of confidence in the performance prediction techniques suffers accordingly. Compensation can be provided by increasing design conservatisms which include increasing heat shield thickness at the expense, of course, of increased weight. This weight increase can become significant depending on the degree of confidence in the prediction techniques. Adoption of design conservatisms as a solution to the problem will be limited, therefore, by weight allowances. Flight test evaluation of the ablation materials is not subject to the same limitations as ground testing and the degree of confidence in the prediction techniques can be improved. The need for flight tests, therefore, hinges on the balance between confidence in the prediction techniques and allowable conservatism. This balance has been judged inadequate and flight tests are recommended.

The recommendation for flight tests was based on:

1. Simultaneous match of stagnation pressure, heat flux, and enthalpy in existing ground test facilities is not possible (but is possible in flight tests).

2. The transient character of the heating environment is difficult to match in ground test facilities.

3. The candidate materials have never been flight tested and hence there is no opportunity to extrapolate, even on a gross basis, a comparison with ground test results.

4. In general, past experience has verified that ground test results can differ from flight test results either in optimistic or conservative directions.

The judgement on the allowable conservatisms is based on the fact that the heat shield weight is 15 to 20 percent of the total entry vehicle weight and any conservatism in its design will represent a significant increase in the entry vehicle weight. This is especially critical when viewed in light of the fact that the vehicle payload weight is only 13 percent of the total and the current margin for growth is 6 percent.

10.1.2 Specific Requirements and Objectives

The flight envelope parameters affecting the heat shield performance (temperature response and mass-loss characteristics) are the entry velocity (V_e), entry angle (γ_e), ballistic coefficient (m/C_DA), and angle of attack (α). Their influence, however, is exerted through the following derivative environmental parameters; aerodynamic heating (Q), heating rate ($\dot{q}$), enthalpy (m/RT_o), pressure (p), shear (τ), duration of the heat pulse, as well as the atmospheric composition.

Design and flight experience, plus consideration of the postulated ablation mechanism for the candidate ablators, indicate that the ablation process is best simulated by providing simultaneous duplication of the heating rates, enthalpy and pressure within reasonable limits. The simultaneous duplication is provided by the selection of the proper combination of the flight envelope parameters: V_e, γ_e, α_e, and m/C_DA. Fortunately, direct simulation of the heating pulse is not necessary because the combined dynamics and heat pulse simulation would be impossible to attain for this particular case. The flight test should be tailored to provide a heat pulse that is only typical of the anticipated heating, with due recognition of thermal protection requirements, booster limitations and instrumentation requirements.

Since entry out of orbit results in low stagnation enthalpies, the entry velocity for the Earth test should be restricted to values near those for Mars to ensure accurate wall enthalpy interactions (other than those introduced by atmospheric composition). A tradeoff between simulation of the total integrated heating and heating rates is possible by variation of the flight envelope, but in any case a transient history similar to Mars entry, is obtained.

Typical Mars entry heating pulses for various stations are illustrated in Figure 30 and for the maximum diameter station in Figure 31. Figure 31 indicates a discontinuous variation in the heating which is associated with the rapid variation in the stagnation point location. In order to simulate this characteristic pulse, the dynamics would have to be simulated. The simulation of the exact heat pulse is not critical for each body station. It is required instead that the heating on the flight test vehicle at a particular body station be related to some point on the Mars entry vehicle, the vehicle scale being compatible with this requirement.

The simulation possible with an Earth entry is demonstrated in Figure 32. Heating rates are presented as a function of stagnation enthalpy, and local pressures are indicated at discrete points. Two points on the body (stagnation point and sonic point) are compared with the corresponding points for the Mars entry. Although there is no one-to-one correspondence of vehicle stations between Earth and Mars entry test vehicles, there is an overlap providing points on the Earth test vehicle which match a region on

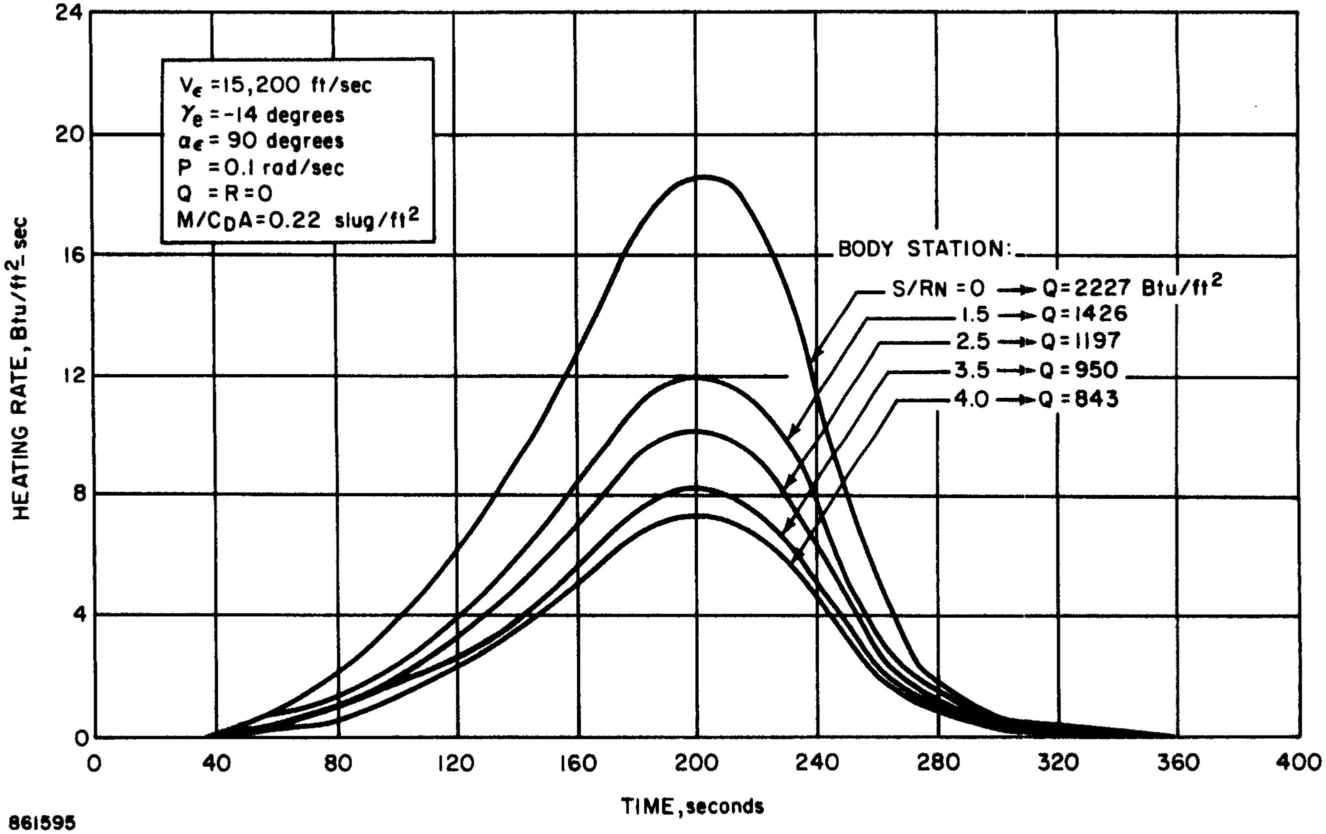

Figure 30 MARS BLUNT CONE HEATING PULSE - VARIOUS BODY STATIONS

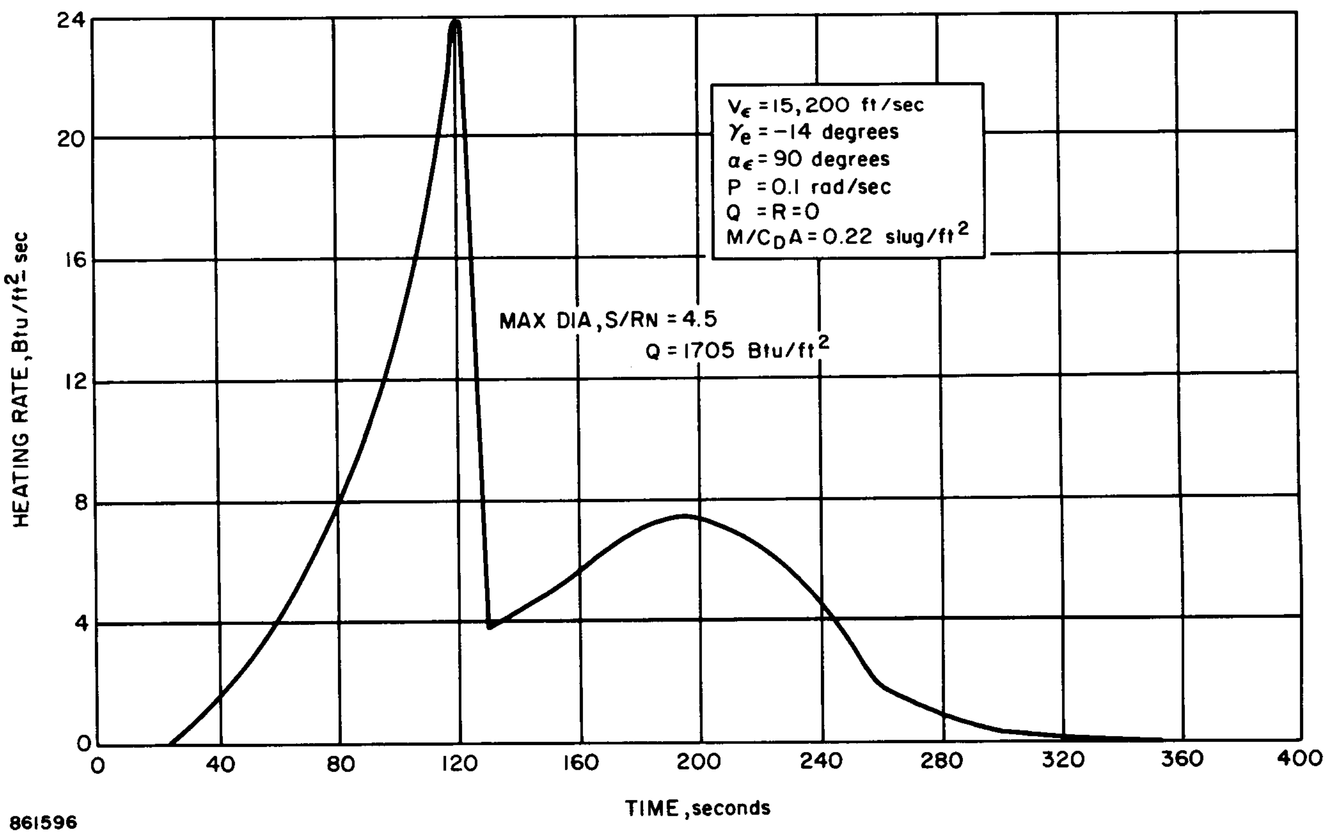

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Figure 31 MARS BLUNT CONE HEATING PULSE AT MAXIMUM DIAMETER

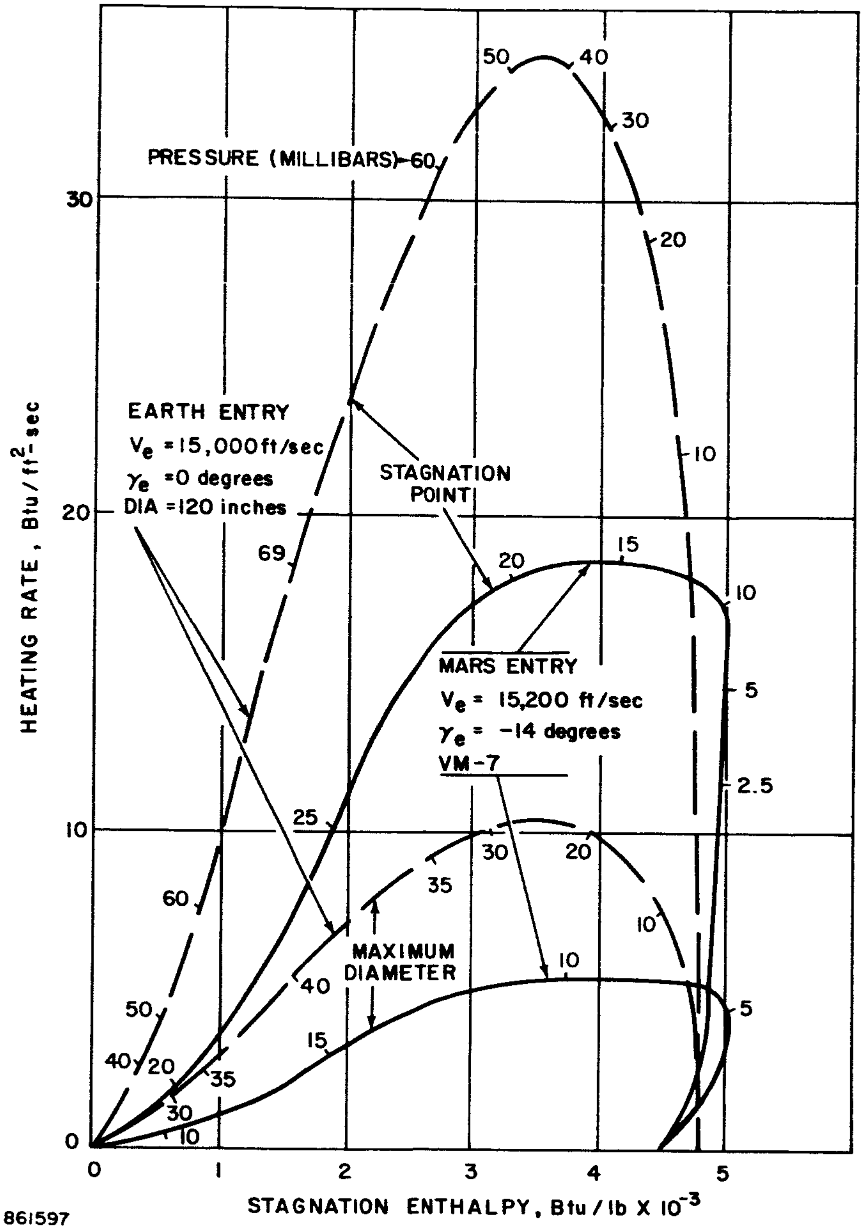

Figure 32 COMPARISON OF MARS HEATING ENVIRONMENT WITH SUBSCALE FLIGHT TEST SIMULATION

the Mars entry vehicle. In such region a simultaneous, transient simulation of the heating rates at appropriate enthalpy levels is possible with small differences in the pressures. A comparison of the Mars and Earth entry heat pulses (Figure 33) shows that simultaneous simulation of the total integrated heating and complete timewise heating-rate distribution is not feasible. It is concluded that an Earth entry test although not executed in the same atmospheric composition as Mars would provide an excellent test of thermal protection system performance. The atmosphere composition effect on performance would have to be demonstrated in the ground tests.

The degree of simulation of the Mars entry in ground test facilities was investigated as well by superimposing their operating characteristics on the Mars entry environmental envelope previously shown in Figure 32. The resulting comparison is presented in Figure 34. This figure demonstrates the difficulty of obtaining low heating rates at the critical (for design) low enthalpy levels, although the range of enthalpies is covered. Furthermore, the pressure simulation is off by an order-of-magnitude which may be important in evaluation of the ablation phenomena. Another well known problem (not illustrated in Figure 34), is the difficulty of simulating timewise variations of the critical parameters in ground facilities. In addition to the proper simulation of the environment, it is important to conduct tests on materials produced to the specification required of the final product, and on a scale approaching the prototype hardware. It is possible in principle to satisfy the material specification but not the scale requirement in ground testing. It is thus concluded that although design information may be acquired in the ground testing, the verification of the performance in these facilities will not provide the degree of confidence in the design that is required.

10.1.3 Aero-Thermodynamic Considerations

Although aerodynamic or aerothermodynamic considerations *per se* do not constitute a flight test requirement, the flight test will provide additional information germane to the aerodynamic performance predictions. As with the heat shield, it is impossible to simulate all the parameters which influence the vehicle performance as well as flow field by means of ground tests. As a valuable adjunct to the heat shield tests, useful data in the form of pressure and heating distributions, as well as vehicle coefficients, can be obtained. The tests will provide data at higher density ratios than are obtained in ground tests. In addition larger variations in flow properties along the boundary-layer edge can be obtained. While the test is primarily tailored for the heat shield, the resulting flight histories and parameter variation (Mach and Reynolds Nos. and density ratio) are good simulations of those anticipated for design purposes.

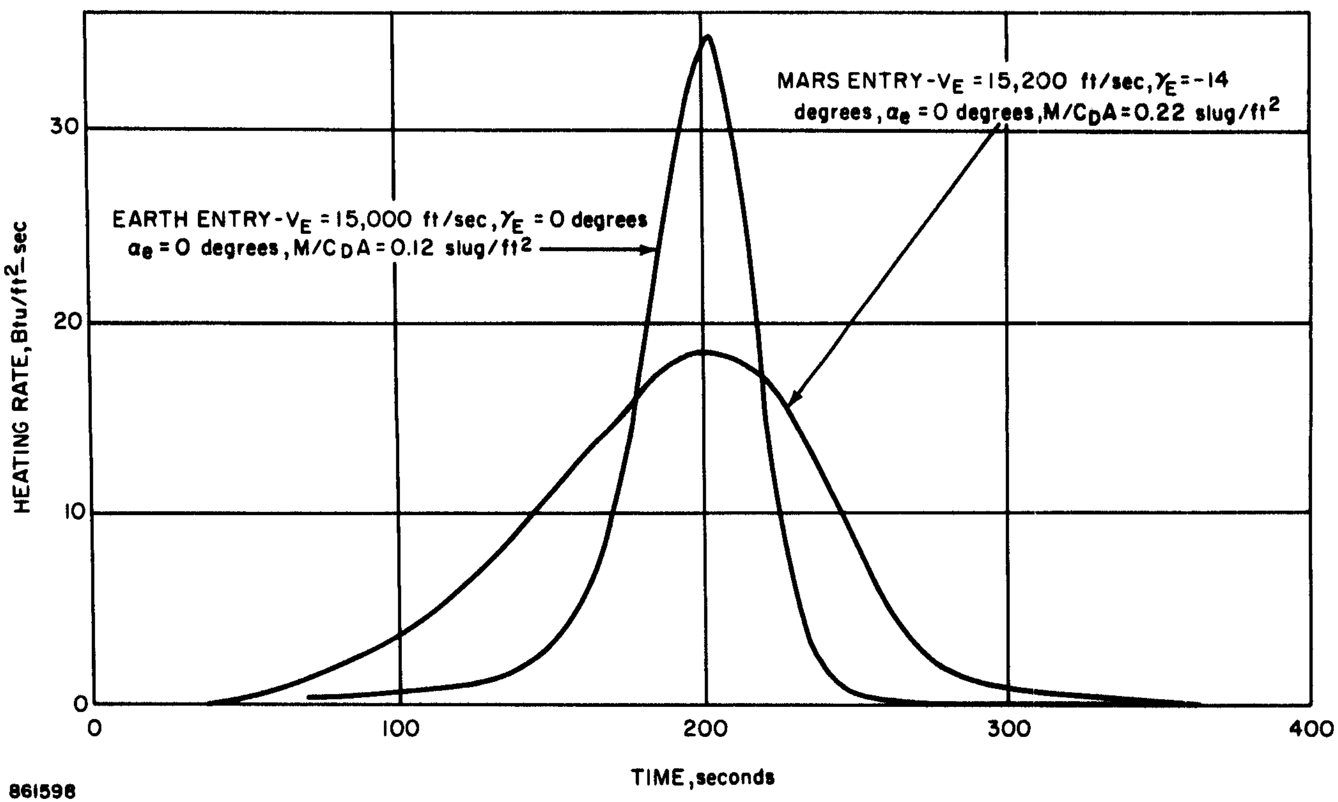

Figure 33 COMPARISON OF MARS AND EARTH ENTRY BLUNT CONE HEATING PULSE

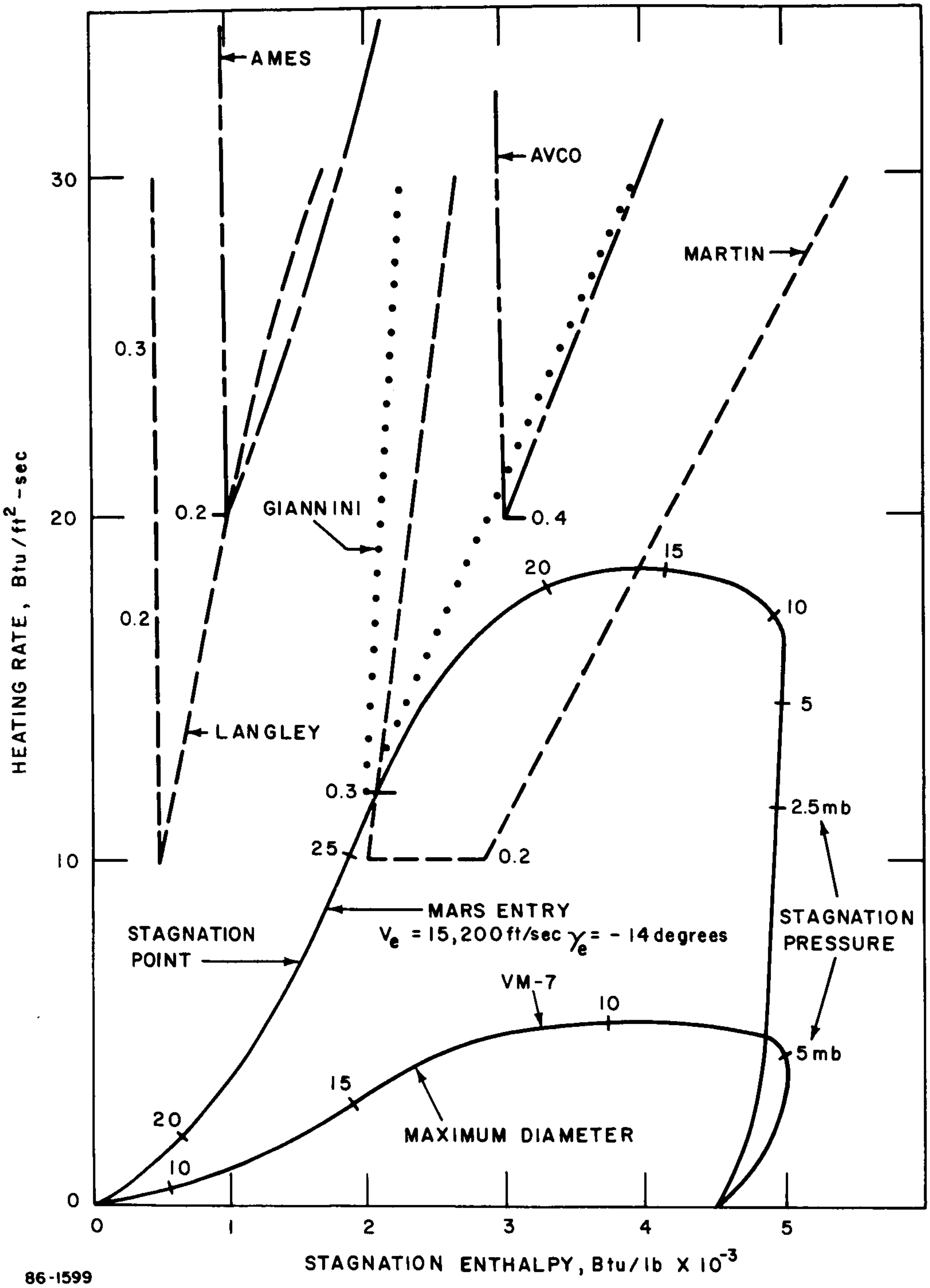

Figure 34 COMPARISON OF MARS HEATING ENVIRONMENT WITH GROUND TEST SIMULATION

The inspection of Mars entry conditions also reveals that an Earth entry test is feasible utilizing a heat-sink thermal protective system. This concept will allow "calibration" of ablator flights and provide definition of the environment unencumbered by ablation products and mass changes. Beryllium must be used for the heat sink to achieve desired entry weights, as demonstrated in Table XXVI.

10.1.4 Scaling

Selection of minimum scaling for the flight test vehicle is very desirable from the viewpoint of minimum launch vehicle capabilities and the associated savings in cost. Unfortunately, significant reduction of the vehicle scale increases the heating flux to the point where adjustment of the other simulation parameters can no longer provide adequate simulation of the Mars heating environment. The studies indicated that the minimum suitable scaling is approximately 100 to 120 inches in vehicle diameter. Figure 32 which was previously discussed shows that a 120-inch diameter vehicle can provide a point on the test vehicle which adequately simulates the Mars stagnation point heating. Further reduction of the test vehicle scale will drive the heating flux at any point on the body beyond the Mars stagnation point heating and eliminate the possibility of simulating the Mars heating environment.

10.2 TEST PROGRAM

The recommended program for the subscale heat shield tests consists of a 100-inch blunt cone test vehicle, launched by an Atlas SLV (OAO) on a direct ascent trajectory, with reentry beginning at ascent burnout. Two test vehicle configurations are required: one with a beryllium heat sink and the other with a heat shield consisting of the material to be evaluated.

Three flights are recommended: one with the heat sink and two with the heat shield. The prime objective of the heat-sink test is to measure the entry heating environment unencumbered by the processes of ablation. All three flights utilize the same reentry conditions: V_e = 15,000 ft/sec and γ_e = 0 degrees, which were selected in combination with the other simulation parameters (vehicle size and ballistic coefficient) to provide the desired heating environment. Two heat shield flights are scheduled to obtain repetitive measurements. The heat shield vehicles are recovered with a parachute and water flotation equipment for post flight examination of the heat shield material. Recovery of the heat-sink vehicle is not required.

As previously explained, the size of the test vehicle could not be significantly reduced below 100 inches in diameter in order to properly simulate the Mars heating. The large vehicle diameter eliminated less costly launch vehicles such as Scout and forced consideration of the Atlas and Titan class of boosters. The Atlas SLV (OAO) was selected because of its availability, its payload interface,

TABLE XXVI

HEAT SINK AND HEAT SHIELD THICKNESS AND WEIGHT FOR EARTH ENTRY TEST

(V_e = 15,000 fps, γ_e = 0° a_e = 0°

(D = 120", $m/C_D A$ = 0.13

Material / $T_{backface}$ / S/R_N	Beryllium(1)	Copper(2)	Purple Blend Mod 5(3)
	500°F	500°F	500°F at bond
0.0	0.85 inches	0.90 inches	0.26 inches
1.5	0.36	0.42	0.18
2.5	0.31	0.34	0.16
4.5	0.27	0.27	0.14
Total Weight (pounds)	257 (113 pounds for 975°F)	1360	49

(1) No substructure required

(2) Substructure required

(3) On reference Mars vehicle honeycomb substructure.

and the similarity between the OAO ascent program and that required for this program. The OAO program is currently scheduled for ten years which hopefully, will ensure the availability of Atlas boosters for the often delayed Voyager program. The payload interface diameter for this booster is 120 inches which is ideal for mating with the 100-inch diameter test vehicle by means of a conical adapter.

The zero-degree reentry angle was an unusual requirement and two launch vehicle trajectories for implementing this angle, were examined. One was a "roller coaster" trajectory, which is accomplished by first reversing pitch attitude after atmospheric exit and firing the sustainer engine downward. After the trajectory turns downward, the pitch attitude is reversed again to thrust upwards such that a pull out maneuver occurs. The objective is to pull out to level flight (γ_e = 0 degrees) at the reentry altitude. The other approach was the one selected. The ascent trajectory is programmed to approach the reentry altitude (400, 000 feet) asymptotically thus providing level flight (or γ_e = 0 degrees). The direct ascent was preferred because it was very similar to an existing ascent program (OAO) and represented a more reliable approach since extreme pitch attitude maneuvers were not required as in the "roller coaster" trajectory.

10.3 LAUNCH VEHICLE CONFIGURATION

The SLV-3(OAO) version of the Atlas with a modified Surveyor ascent shroud is recommended for this test. This version of the Atlas has a 120-inch diameter interface with the payload at the forward tank ring. The OAO fixed adapter which is a cylindrical section 30 inches long mates with the forward tank ring. The forward adapter ring mates with a conical adapter for the test vehicle and the ascent shroud in a three-way joint. Since the overall length of the test vehicle is very short, the Surveyor shroud is shortened by removing its cylindrical section and retaining only the conical section. Since the test vehicle is separated from the Atlas at the beginning of reentry, retropropulsion of the Atlas may be necessary. A cluster of solid rockets would be added for this purpose. The ascent shroud weight is 1000 pounds, the payload conical adapter is 200 pounds, the OAO fixed adapter 320 pounds, and the test vehicle weight is 500 pounds for a total of 2020 pounds. This total payload weight is well within the capability of the Atlas for the suborbital performance required.

10.4 TEST VEHICLE CONFIGURATION

The heat shield flight test vehicle is a 100-inch diameter (approximately half-scale) boilerplate mockup which simulates the external aerodynamic shape of the operational prototype. Two versions are required: one which utilizes a heat sink for thermal protection and the other which employs the heat shield material which is to be evaluated. Each vehicle weighs 500 pounds. Ballast is added to the heat shield vehicle to match the heat sink vehicle weight. The internal structure does not simulate the prototype configuration. Its design is functional, providing support for subsystems and components in a configuration which will yield a favorable c. g. location. No separations are utilized except for deployment of the recovery parachute in the heat shield material tests. An inboard profile of the vehicle, mounted on its launch vehicle, is shown in Figure 35.

The important subsystems and components are tabulated below:

1. Heat shield (or heat sink)
2. Entry vehicle shell
3. Internal structure
4. Atlas/test vehicle adapter
5. Atlas/test vehicle separation system
6. FM/FM Telemetry
7. Battery Power supply
8. Programmer and control circuitry
9. Pressure sensors**
10. Calorimeters*
11. Ablation gages*
12. Thermocouples
13. Accelerometers
14. Rate gyros
15. Recovery parachute*
16. Foam buoyancy material*
17. C-band tracking beacon
18. SARAH beacon*
19. Spin rockets
20. Tape recorder

*Excluded from heat-sink test

**Excluded from heat shield material test

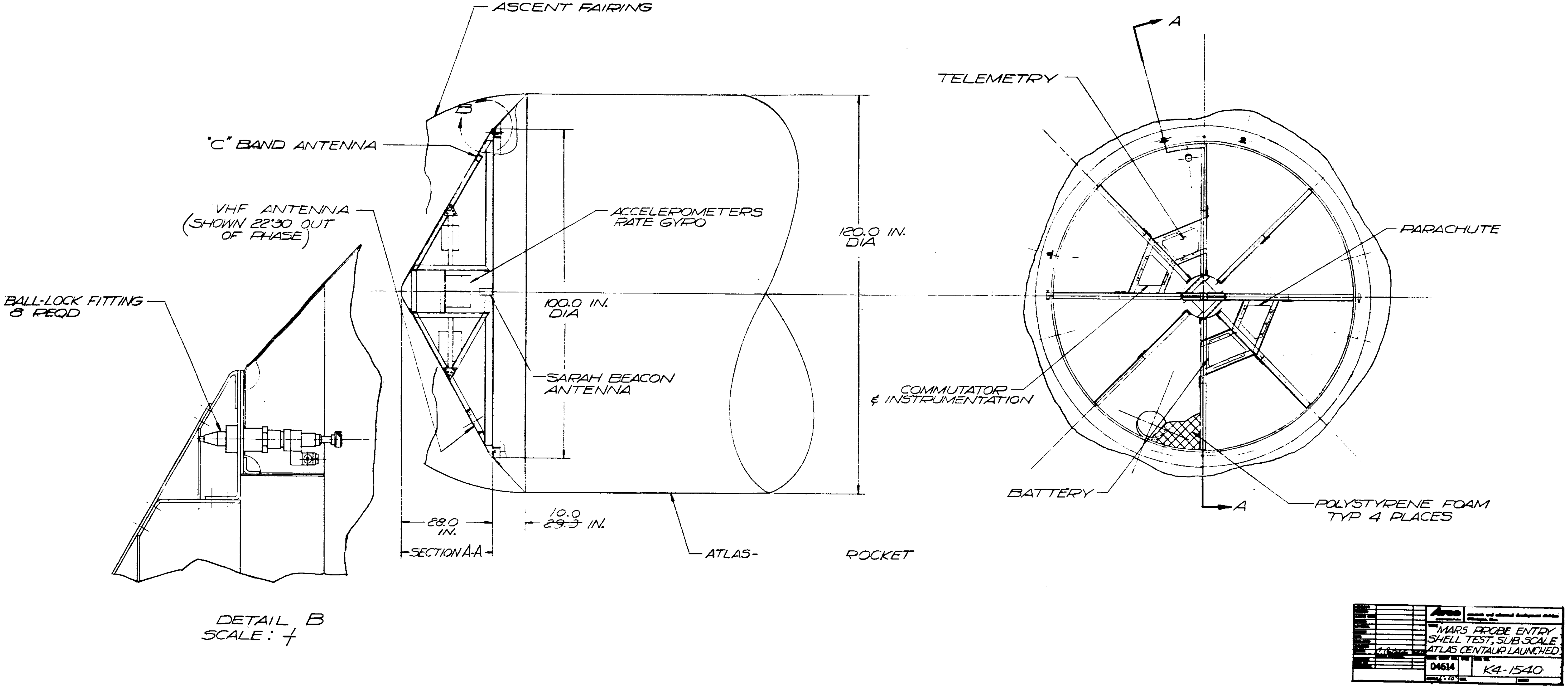

Figure 35 INBOARD PROFILE OF SUBSCALE HEAT SHIELD TEST VEHICLE -- ATLAS SLV-3 (OAO) LAUNCH

The entry vehicle shell is a monocoque shell stiffened against buckling by a skeleton frame of angles and a fabricated circumferential ring at the maximum diameter. The heat shield material is bonded to the monocoque shell. The beryllium heat sink vehicle does not require support by a monocoque shell, its thickness and strength are adequate. The internal structure and circumferential ring are riveted directly to the beryllium heat sink. The ring mates with a ring-stiffened conical adapter which transmits the evenly distributed launch loads to the Atlas interface. The internal structure is a truss assembly welded to the shell frame. The subsystems are mounted on this truss as shown in the inboard profile.

The separation system consists of eight ball-lock mechanisms evenly distributed around the circumferential interface with the launch vehicle adapter. Spin rockets are mounted on the outer ring of the vehicle. It is not certain that vehicle spin stabilization is required since the vehicle is released from the Atlas very close to the sensible atmosphere and at zero-angle of attack. A detailed dynamic analysis will be required to resolve this question.

For the heat shield tests, recovery and postflight examination of the heat shield is important. The recovery system consists of a descent parachute, low density polystyrene foam for buoyancy, C-band tracking beacon, SARAH recovery beacon and dye markers.

The heat sink and heat shield instrumentation sensors such as pressure sensors, calorimeters, ablation gages, and thermocouples will be distributed in patterns to give both meridional and longitudinal coverage. Data during blackout will be tape recorded and replayed over several cycles after exit from blackout as indicated by a timer. Accelerometers will be installed at the vehicle c. g. as indicated in the inboard profile.

10.5 FLIGHT SEQUENCE

The flight sequence for the subscale heat shield test is illustrated in Figure 36. The zero-degree reentry angle is an unusual requirement but is easily implemented by terminating the powered ascent trajectory at the reentry conditions, without the usual long range ballistic flight between burnout and reentry. The ascent trajectory is similar to that required for the low altitude orbit of the OAO satellite. The desired ascent trajectory is implemented by a pitch rate program during sustainer engine burn. The ascent shroud is jettisoned at 300,000 feet or higher during sustainer engine burn. The pitch rate program provides an asymptotic approach to the reentry altitude, 400,000 feet. The reentry conditions (V_e = 15,000 ft/sec and γ_e = 0 degrees) are achieved at 400,000 feet. The sustainer engine is cutoff, the entry vehicle separated, and the solid rocket retromotors ignited, in that order. Angle of attack at separation is zero degrees. No vernier engine is required because the accuracy of the sustainer engine cutoff is more than adequate for this mission. Since separation occurs close to the sensible atmosphere and at zero-angle of attack, spin

The entry vehicle shell is a monocoque shell stiffened against buckling by a skeleton frame of angles and a fabricated circumferential ring at the maximum diameter. The heat shield material is bonded to the monocoque shell. The beryllium heat sink vehicle does not require support by a monocoque shell, its thickness and strength are adequate. The internal structure and circumferential ring are riveted directly to the beryllium heat sink. The ring mates with a ring-stiffened conical adapter which transmits the evenly distributed launch loads to the Atlas interface. The internal structure is a truss assembly welded to the shell frame. The subsystems are mounted on this truss as shown in the inboard profile.

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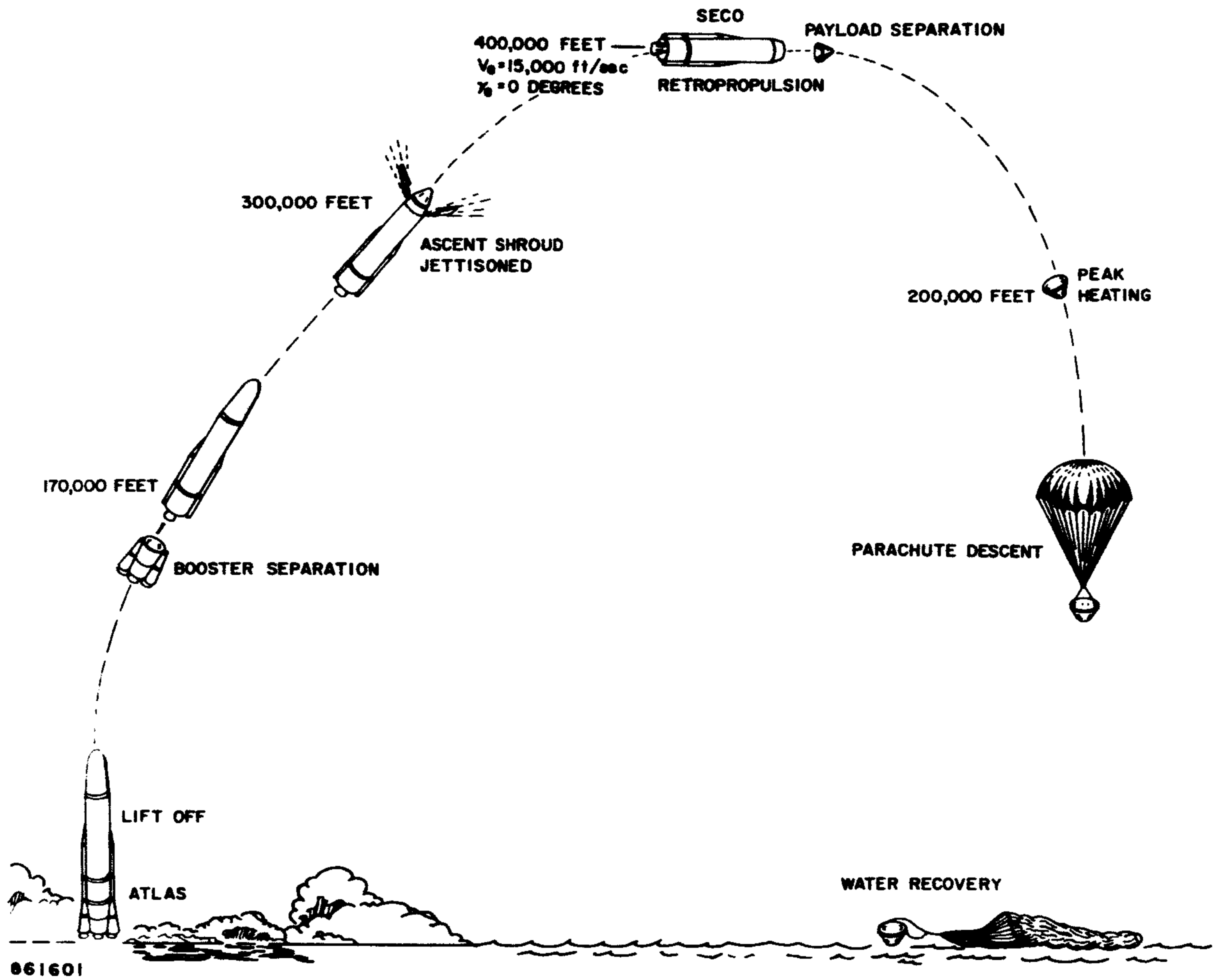


Figure 36 SUBSCALE HEAT SHIELD FLIGHT TEST - ATLAS SLV-3 (OAO) LAUNCH-FLIGHT SEQUENCE

stabilization of the vehicle may be unnecessary. Peak heating of the vehicle will occur near 200, 000 feet. During blackout, data is tape recorded and re-transmitted later. Data is telemetered in real time both before and after blackout. After reaching terminal velocity, the recovery parachute is deployed to recover the heat shield for postflight examination. The heat-sink model is not recovered.

APPENDIX A

<u>PRELIMINARY PROGRAM PLANNING FOR 1971 PROBE/LANDER MISSION</u>

1.0 INTRODUCTION

During the course of the Part 1 studies, new information relating to the characteristics of the Martian atmosphere became available leading to a redirection of the study effort by NASA. Included with the redirection was the deletion of the requirement for a program plan.

This appendix contains the program planning activities for a 1971 Probe/Lander mission accomplished during Part 1. It is included as a typical plan, although not directly applicable to the study effort performed during Part II.

A primary goal during Part I was the formulation and establishment of a format and guidelines to be used during Part II for the preparation of a detail program plan. This appendix contains the assumptions, the work breakdown structure, and the hardware and schedule used in establishing the baseline for estimating resources requirements.

2.0 MASTER PHASING CHART

Figure A-1 shows the Master Phasing Chart developed to provide broad program visibility of the interrelationships and duration of the major activities for a 1971 Probe/Lander mission. The schedule is based on a number of assumptions that relate to program activities and available technical techniques.

2.1 ASSUMPTIONS

Certain assumptions relating to the form and details of the program were established during the early phase of this study. Together with the system performance requirements and the mission objectives, these assumptions formed the basis for the development of the preliminary work breakdown structure, schedule, and cost.

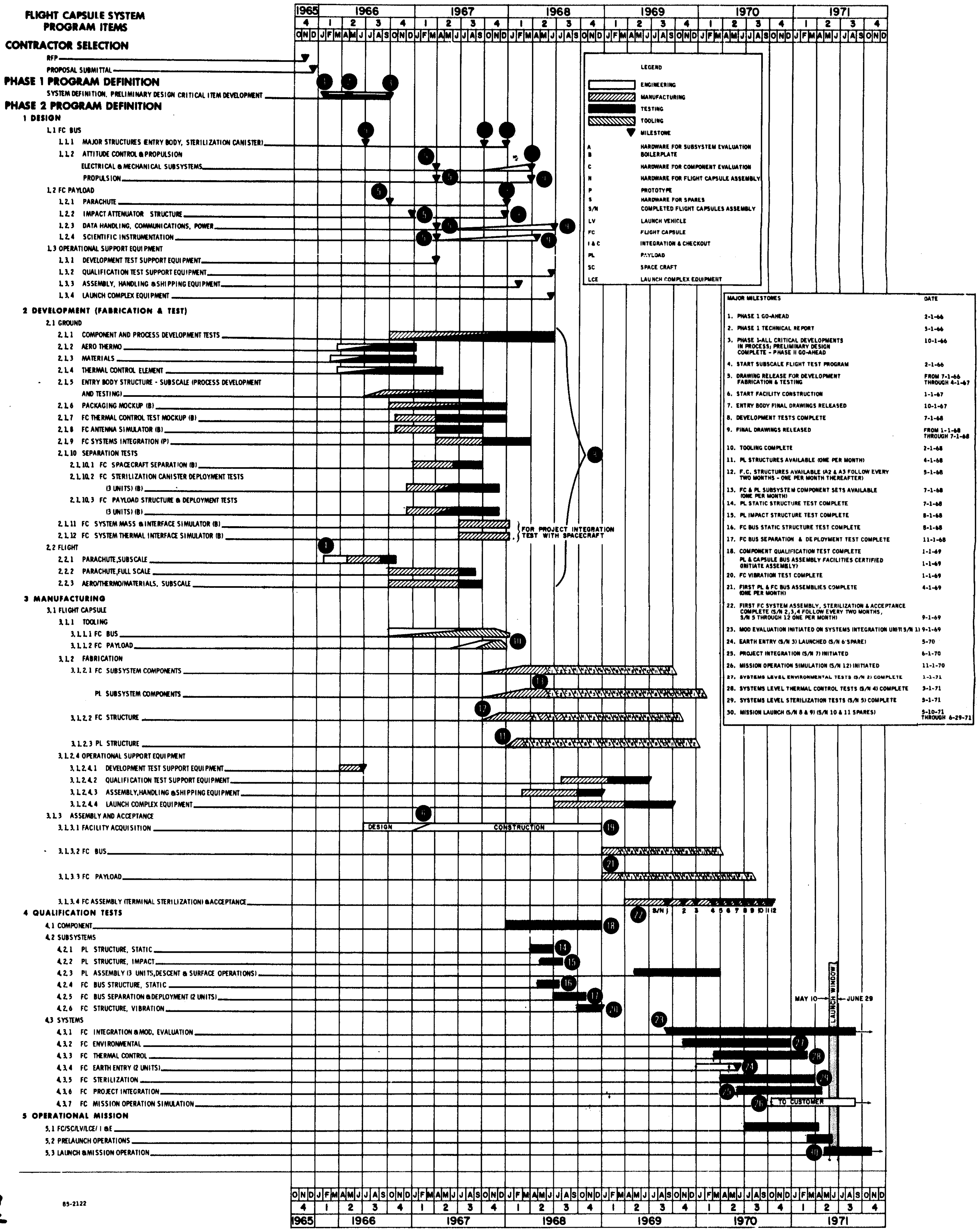
2.1.1 Programming

A program will be initiated in February 1966 for the development of the Voyager flight capsule system.

Voyager missions will occur in 1971, 1973, 1975, and 1977.

There will be no gap in preliminary design efforts in 1966 in the selection process for the flight capsule contractor for the hardware contract.

Critical earth entry test programs for parachute development and for the verification of the aerodynamic-thermodynamic performance of the entry



174-2

Figure A-1 MASTER PHASING CHART FOR 1971 PROBE/LANDER MISSION shell will be initiated in early 1966. This scheduling is necessary in order to properly factor the results into the design cycle for the flight capsule.

A full-scale flight capsule earth entry system test is scheduled in mid-1970. Earlier scheduling is desirable but not possible if ground qualified hardware is to be used for this test.

2.1.2 Design

A single, large multi-mission vehicle shell, 15 to 16 feet in diameter, will be used for all missions, with only minor modifications required in the shell design.

The 1971 mission will utilize a parachute descent system and a hardlanded payload protected by passive impact energy absorbers.

RTG's will not be used in 1971 or 1973.

2.1.3 Sterilization and Manufacturing

The basic techniques for meeting the NASA planetary quarantine specifications will be that of viable organism population control (burden control) during assembly, encapsulation of the flight capsule within a rigid sterilization container, and dry-heat terminal sterilization of the flight capsule within its sterilization container.

The manufacture of parts and components will take place in conventional facilities.

Payload assembly will be performed in Class 100 facilities per Federal Specification 209.

Major structural elements will be of conventional manufacture but are surface-cleaned and treated with ETO prior to final assembly and terminal sterilization.

Final assembly and acceptance will be conducted at Cape Kennedy in accordance with NASA's direction.

2.1.4 Flight Qualified Hardware

Two flight capsules will be launched in 1971; each from a separate launch pad.

Two spare flight capsules will be provided as backup for the 1971 launch.

Each complete flight capsule set (flight articles plus spares) will have a spare payload assembly available for substitution.

Each set of components and subsystems will have 100 percent spares available for substitution during assembly.

2.2 CRITICAL SCHEDULE ACTIVITIES

The events whose activities are considered critical to the schedule success are those relating to the development and conduct of the flight test programs. Recognizing this, considerable attention was given early in the study to the requirements for development of flight tests.

It will be necessary to initiate the planning and test article system design very early in the program. This will permit the acquisition of the data necessary to refine the design prior to its release for systems level demonstrations and mission operation use.

If a combination of subscale and full-scale high-altitude rocket tests of the parachute be selected, early prosecution of the subscale tests is recommended. These early tests are not mandatory but are recommended for extra insurance against unexpected development problems. The subscale parachute is the only one that can be executed as early as shown because it utilizes a test payload that only simulates mass. The other tests employ mockups of the capsule and its subsystems in which mass, geometry, and functions are simulated, and hence must await completion of at least the flight prototype system design and subsystem functional specifications. In order to facilitate an early flight test schedule for these items, flight test planning, test article system design, and subsystem functional specifications should proceed simultaneously with the same developments for the flight prototype during Phase I. Detail design of the subsystems and subcontracting can then begin at the initiation of Phase II. Approximately nine months are allowed for completion of the detail design, fabrication, assembly, and checkout for the full-scale parachute tests and the aerothermo ballistic reentry tests. This will permit completion of the flight tests approximately 3 months prior to final drawing release for the test items.

A full-scale Earth entry test is scheduled in mid-1970. It would be desirable to schedule this test earlier, but since ground-qualified hardware is required for this test, earlier scheduling is not feasible.

3.0 WORK BREAKDOWN STRUCTURE

The work breakdown structure (WBS) identified (through successive stages of subdivision) the hardware, services, functions, support equipment, and facilities required in the performance of a program. It provides the basis for:

1. Uniform planning and program visibility

2. Assignment of responsibilities

3. Monitoring progress

4. Networking

5. A framework for achieving cost visibility

6. Defining required end items of equipment, services, and facilities.

Figure A-2 depicts a version of the WBS developed during Part I of the study. The major areas of work identified are:

Project Controls

Procurement/Production Management

Product Assurance

Systems Integration

Flight Capsule - Development

Flight Capsule - Qualification

Flight Capsule - Operational/Flight Units

Operational Support Equipment - Development

Operational Support Equipment - Qualification

Operational Support Equipment - Deliverable

These are all identified at "level 2" on the WBS and represent the initial subdivision of the total program. Continued subdivision of the above work areas and hardware into lower levels will define the program work requirements and the contract end items more accurately. Having identified the contract end items, work packages which define specific activities and functional responsibilities may then be established. At this level, detailed schedule and cost information are developed.

Prior to NASA's redirection of the programs effort, Part 2 of the study effort was to involve the detailed subdivision of the WBS, the identification of "contract" end items and their major components, and the assignment of a coding scheme to WBS elements for cost collection and monitoring purposes. Work packages

PROGRAM BREAKDOWN STRUCTURE
CAPSULE SYSTEM

LEVEL 1	2	3	4
Capsule System	Project Controls	Financial & Schedule Control	
		Data Management	Program Plans
			Documentation
		Contract Management	
		Subcontract Management	
		Configuration Management	Configuration Control
			Configuration Identification
			Configuration Accounting
			Records and Standards
	Procurement/Production Management	Administration	
		Material Management	Procurement
			Stores
			Inventory Control
		Tooling & Process Engineering	
		Tool Fabrication	

Figure A-2 PROGRAM BREAKDOWN STRUCTURE CAPSULE SYSTEM

LEVEL 1	2	3	4
	Product Assurance Management	Reliability Assurance	Reliability Engineering
			Failure Analysis
			Corrective Action
			Safety Assurance
			Administration
		Quality Assurance	Quality Engineering
			Quality Administration
			Inspection and Test
			Vendor Control
		Sterilization	Methods and Controls
			Laboratory
		Data Bank	
	Systems Integration	Systems Analysis and Definition	Systems Definition and Control
			Systems Assurance
			Systems and Operational Analysis
			Advanced Studies

Figure A-2 PROGRAM BREAKDOWN STRUCTURE CAPSULE SYSTEM (Cont'd)

LEVEL 1	2	3	4
		Design Integration	AVE Integration
			Science Integration
			OSE Integration
			Interface Controls
		Systems Test Management	Integrated Test Planning
			Field Testing
			Laboratory Testing
			TSE Development
			Mission Operations
			Analysis
	Flight Capsule Development	Flight Spacecraft to Flight Capsule Adapter	Structure Assembly Separation Subsystem Cabling
		Sterilization Canister	Structure Assembly Separation Subsystem Pressurization Subsystem
		Capsule Science Payload	
		Capsule Bus	Entry Shell
			Thermal Control
			Separation and Deployment
			Propulsion
			Parachutes

Figure A-2 PROGRAM BREAKDOWN STRUCTURE CAPSULE SYSTEM (Cont'd)

LEVEL 1	2	3	4
		Capsule Bus (cont'd)	Impact Attenuation
			Attitude Control
			Signal/Power Interconnect
			Programming/Sequencing
			Telecommunications
			Electric Power and Control
		Flight Test Instrumentation	
		Models	Packaging Model
			Thermal Control Model
			Antenna Model
			Flight Test Model
			Wind Tunnel Model
	Flight Capsule Qualification	Qualification Tests	Flight Test Program
			Ground Test Programs
		Flight Capsule for Qualification Test	Flight Spacecraft to Flight Capsule Adapter
			Sterilization Canister
			Capsule Science Payload
			Capsule Bus
			Flight Test Instrumentation

Figure A-2 PROGRAM BREAKDOWN STRUCTURE CAPSULE SYSTEM (Cont'd)

LEVEL 1	2	3	4
	Flight Capsule - Operational Flight Units	Flight Spacecraft to Flight Capsule Adapter	
		Sterilization Canister	
		Capsule Science Payload	
		Capsule Bus	Entry Shell
			Thermal Control
			Separation and Deployment
			Propulsion
			Parachutes
			Impact Attenuation
			Attitude Control
			Signal/Power Interconnect
			Programming/Sequencing
			Telecommunications
			Electric Power and Control
	Operational Support Equipment - Development	Facilities	
		Assembly, Handling and Shipping Equipment	
		Systems Test Complex	Electrical
			Mechanical

Figure A-2 PROGRAM BREAKDOWN STRUCTURE CAPSULE SYSTEM (Cont'd)

LEVEL 1	2	3	4
		Launch Complex Equipment	Electrical
			Mechanical
		Mission Dependent Equipment	Electrical
			Mechanical
	Operational Support Equipment - Qualification	Qualification Tests	Component Qualification Tests
			Subsystem Qualification Tests
			Systems Qualification Tests
		Operational Support Equipment for Qualification Tests	Assembly, Handling and Shipping Equipment
			Systems Test Complex
			Launch Complex Equipment
			Mission Dependent Equipment
	Operational Support Equipment - Deliverable	Assembly, Handling and Shipping Equipment	
		Systems Test Complex	Electrical
			Mechanical
		Launch Complex Equipment	Electrical
			Mechanical
		Mission Dependent Equipment	Electrical
			Mechanical

Figure A-2 PROGRAM BREAKDOWN STRUCTURE CAPSULE SYSTEM (Concl'd) would then have been identified and detailed schedule networks and cost data developed. Integrating the schedule and cost data developed at each level of the WBS would provide total schedule and cost requirements for the program.

4.0 HARDWARE REQUIREMENTS

Figure A-3 is a preliminary identification of the levels, types, and quantities required to satisfy and support the development, qualification, and operational needs for a sterilized flight capsule, within the period of performance shown on the master phasing chart. The table is arranged to present the preliminary hardware types and quantities needed to support the program requirements for development, qualification, and mission operation. Figure A-4 presents the summary of hardware usage and the estimated completion dates for the qualification and flight hardware article.

4.1 FLIGHT QUALIFIED HARDWARE

The total quantities of flight qualified hardware, including spares, required to support the 1971 launch opportunity was given in paragraph 2.1.4.

4.2 DEVELOPMENT AND QUALIFICATION HARDWARE

The quantities shown for the development and qualification program reflect schedule constraints which preclude the use of the same test article for more than one type of development or system testing.

Prior to NASA's redirection, Part II of the study would have entailed a refinement of the detail requirements to support the development and qualification of the selected system. This would have involved further examination of the state-of-the-art of the hardware to be used and the necessary development and qualification procedures. Hardware needs versus schedule requirements would have been investigated in detail to determine whether the use of a test article for many tests can be made compatible with the schedule, thereby reducing hardware.

	F. C. Bus Structure			F. C. Bus Subsystem Components			Payload Structure			Payload Subsystem Components			F.C. Bus (Structure and Subsystem Comp.)			Payload Assembly (Structure and Subsystem Comp.)			Flight Capsule		
No.	Code*	Completion Date	Usage	Code*	Completion Date	Usage	Code*	Completion Date	Usage	Code*	Completion Date	Usage	Code*	Completion Date	Usage	Code*	Completion Date	Usage	Code*	Completion Date	Usage
1	A1	5/1/68	Bus structure static test	C1	7/1/68		A1	4/1/68	P. L. structure static test	C1	7/1/68		N1	4/1/68		N1	4/1/69	S/N1	S/N1	9/1/69	Integration and mod. evaluation
2	A2	7/1	Bus separation & deployment test	C2	8/1	Component qualification tests	A2	5/1		C2	8/1	Component qualification tests	N2	5/1		A1	5/1	Descent and surface operation test	S/N2	11/1	Environmental tests
3	A3	9/1		C3	9/1		A3	6/1	P. L. structure impact tests	C3	9/1		N3	6/1		N2	6/1	S/N2	S/N3	1/1/70	Earth entry test
4	A4	10/1	Vibration test	N1	10/1		A4	7/1		A1	10/1	Descent and surface operation tests	N4	7/1		A2	7/1	Descent and surface operation test	S/N4	3/1	Thermal control tests
5	N1	11/1		N2	11/1		A5	8/1	Vibration test	A2	11/1		N5	8/1		N3	8/1	S/N3	S/N5	4/1	Sterilization tests
6	N2	12/1		N3	12/1		A6	9/1	Descent and surface operation tests	A3	12/1		N6	9/1	F.C. assembly	A3	9/1	Descent and surface operation test	S/N6	5/1	Earth entry test (spare)
7	N3	1/1/69		N4	1/1/69		A7	10/1		N1	1/1/69		N7	10/1		N4	10/1	S/N4	S/N7	6/1	Project integration tests
8	N4	2/1		N5	2/1		A8	11/1		N2	2/1		N8	11/1		N5	11/1	S/N5	S/N8	7/1	Launch
9	N5	3/1		N6	3/1	F. C. bus assembly	N1	12/1		N3	3/1		N9	12/1		N6	12/1	S/N6	S/N9	8/1	Launch
10	N6	4/1	F. C. bus assembly	N7	4/1		N2	1/1/69		N4	4/1		N10	1/1/70		N7	1/1/70	S/N7	S/N10	9/1	Launch spare
11	N7	5/1		N8	5/1		N3	2/1		N5	5/1		N11	2/1		N/8	2/1	S/N8	S/N11	10/1	Launch spare
12	N8	6/1		N9	6/1		N4	3/1		N6	6/1	P. L. assembly	N/12	3/1		N/9	3/1	S/N9	S/N12	11/1	Mission operation tests
13	N9	7/1		N10	7/1		N5	4/1		N7	7/1		S1	4/1	F.C. bus spare	N10	4/1	S/N10			
14	N10	8/1		N11	8/1		N6	5/1	P. L. assembly	N8	8/1					N11	5/1	S/N11			
15	N11	9/1		N12	9/1		N7	6/1		N9	9/1					N12	6/1	S/N12			
16	N12	10/1		S1	10/1		N8	7/1		N10	10/1					S1	7/1				
17	S1	11/1	Spare				N9	8/1		N11	11/1					S2	8/1	P. L. assembly spares			
18							N10	9/1		N12	12/1										
19							N11	10/1		S1	1/1/70										
20							N12	11/1		S2	2/1	Spares									
21							S1	12/1	Spares												
22							S2	1/1/70													

*Master phasing
chart identification

Note 1: This summary does not include development hardware requirements

Figure A-4 VOYAGER CAPSULE PROGRAM--HARDWARE USAGE SUMMARY

ITEM / (1) USAGE	Flight capsule	Capsule Bus	Structure	Subsystem Component	Payload	Structure	Subsystem Component	Operational Support Equipment
1. DEVELOPMENT TESTS								
1.1 Ground	6	4		5	3		5	3
Breadboard components				5			5	3
Packaging mockup	1							
Thermal mockup	1							
Antenna simulator	1							
System integration unit	1							
Flight capsule - spacecraft separation		1						
Canister deployment		3						
Payload deployment					3			
Mass & interface simulator	1							
Thermal interface simulator	1							
1.2 Flight (2)		3				6	6	
Parachute (sub-scale) test model						4	4	
Parachute (full-scale) test model						2	2	
Aero-thermo/materials (sub-scale) - test model		3						
2. QUALIFICATION TESTS	9		3	3	3	4	3	3
Components				3			3	3
Payload structure, static						1		
Payload structure, impact						3		
Payload deployment					3			
Capsule bus structure, static			1					
Capsule bus deployment			2					
*Flight capsule system vibration	1							
System integration & Mod. Evaluation	1							
Environmental	1							
Thermal Control	1							
Earth entry	2							
Sterilization	1							
Project integration	1							
Mission operation simulator	1							
3. MISSION OPERATION	4	1			2			3
1971 Launch window	2							2
Spares	2	1			2			1

Notes:

(1) Quantities shown are at the level of usage, and consist of all equipment making up the respective assembly.

(2) Sparing not included.

* Dummy loads substituted for subsystems components.

Figure A-3 HARDWARE REQUIREMENTS